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Lewis Flight Propulsion Laboratory
Cleveland, Ohio

November 22, 1957

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November 22, 1957

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INTRODUCTION

This volume contains copies of the technical papers which are considered to be Restricted Data and were presented at the "NACA 1957 Flight Propulsion Conference," held at the Lewis Flight Propulsion Laboratory on November 22, 1957. A list of those attending the conference is included.

The original presentation and this record are considered supplementary to, rather than substitutes for, the Committee's system of complete and formal reports

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1. NUCLEAR LOGISTIC CARRIER

By Paul G. Johnson, James W. Miser, and Roger L. Smith

This discussion of air-breathing nuclear propulsion systems is limited to their use in large, medium-altitude aircraft. Specifically, the study involves turbojet aircraft of 500,000-pound gross weight designed for flight at 35,000-feet altitude in the Mach number range from 0.9 to 2.5. Three types of nuclear-propulsion systems are presented for comparison: (1) a direct air system, (2) a liquid-metal system using lithium 7 as the reactor coolant, and (3) a helium system. All shields are "unit shields."

A word of caution must be emphasized at this point. A rough comparison of these three cycles is presented herein for a very restricted design-point range based on calculations involving many assumptions. Any conclusions drawn from the figures must be applied only to these conditions. The relative merits of the three cycles presented for logistic carriers cannot be generalized to other flight conditions or aircraft missions.

Schematic diagrams of the three propulsion systems and some of their advantages and disadvantages are given in figure 1. In all the systems several turbojet engines would be run with one reactor. In the direct air system (fig. 1(a)) the air leaving the turbojet compressor would be ducted to the reactor, heated in passing over the fuel elements, and ducted back to the turbine. The advantages and disadvantages of this system are listed in figure 1(a). Since air is a relatively poor heat-transfer fluid, even at the pressures resulting from high flight Mach numbers and compressor pressure ratios, the reactor core will be relatively large. The reactor shield will be correspondingly heavy, especially when it is a unit shield. An offsetting advantage of the direct air system, other than simplicity, is the elimination of intermediate heat-transfer processes, with the temperature drops and auxiliary pumping power requirements that

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accompany them. The major obstacle standing in the way of achieving high turbine-inlet temperature with the air cycle is oxidation of the fuel elements. It is not the intention in this paper to predict the extent to which the oxidation limit can be pushed back through materials research; instead, the temperature required to achieve a certain performance level will be estimated.

The lithium system is shown in figure 1(b). In this system the air of the turbojet is heated in a heat exchanger located between the compressor and turbine. The lithium circulates in a closed loop transporting the reactor heat to the engine heat exchanger. The pump to circulate the lithium is driven off the turbojet shaft. It is assumed that the isotope separation of the lithium will be sufficient to make an intermediate heat exchanger unnecessary. In other words, the lithium coolant will be so predominantly lithium 7 that the cross section of the mixture will be extremely low, and no activation problem will be encountered. Thus, the lithium that cools the reactor can be brought outside the shield without complicating the shielding problem. Liquid metals are very good heat-transfer fluids, and lithium is the best of these with respect to thermodynamics. As a result, the reactor core of this system is relatively small and the shield is correspondingly light. Temperature differences between the lithium and either the reactor fuel elements or the heat-exchanger wall will be small. The major problem in any liquid-metal system is corrosion and mass transfer. Very little work has been done on lithium-containment at high temperatures, but recent tests at Pratt & Whitney indicate good compatibility with columbium at 1500° F. Unfortunately, columbium oxidizes very readily in air and thus is not a heat-exchanger material. Some type of bimetal construction will be necessary.

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Figure 1(c) shows the system that uses high-pressure helium as the reactor coolant. The schematic diagram is identical to the previous slide of the lithium system. The coolant fluid has changed, but it is still used only as a heat-transfer medium. Variations using helium turbines are possible, but helium is such a poor working fluid that the turbomachinery required would be quite heavy. Because of the high pressure of the helium (about 1700 psi) the lines and heat exchanger will be heavy, but this pressure is an optimum compromise between these high component weights and shield weight, which decreases as the high pressure reduces the reactor core size. The principal reason for using helium as the reactor coolant is its chemical inertness. Although oxidation of the air side of the heat exchanger is still as much of a problem as ever, the problem area is removed from the reactor to a point where it seems less formidable. Reactor fuel elements of molybdenum could be expected to stand much higher temperatures than would result in an oxidation problem in the heat exchanger. Thus, the critical temperature in the helium system is the maximum metal temperature in the heat exchanger, which is limited by oxidation or stress-rupture. Helium is also a convenient coolant, because it does not become radioactive. As in the lithium cycle, no intermediate heat exchanger is required.

At 1700 pounds per square inch, the density of the helium entering the reactor is approximately equal to the density of the air entering the reactor of the direct air system. Since the helium has a specific heat nearly five times that of air, the required coolant volume in the reactor can be reduced by this factor at least. A large difference in sonic speed between helium and air also favors the use of helium because of the resulting difference in pressure drop.

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Lithium, by these criteria, is a great improvement over either helium or air. However, as shown in figure 2, the large gains occurred in the change from air to helium, and further improvement is small because reactor size becomes more a function of criticality than of required coolant volume. The curve of reactor diameter against reactor volume occupied by coolant is determined from criticality calculations. Presence of the coolant has been ignored in plotting the curve. The points plotted are for the three coolant fluids removing 700 megawatts of reactor power. The points correspond to the optimum reactors from the study at Mach 2 at an altitude of 35,000 feet. Two conclusions can be drawn from this plot: (1) For this high reactor power, the direct air system will require either a very large reactor or more than one reactor with associated shield weight penalty; and (2) for this particular set of conditions, helium is nearly as effective as lithium in reducing reactor size and shield weight.

But a comparison based on an individual powerplant component does not tell the whole story. A better indication of the relative merits of the three systems is given by an integration of all components into a comparison of aircraft performance. In this study the design-point performance of 500,000-pound-gross-weight aircraft is compared in terms of payload at various flight Mach numbers and altitudes. No allowance is made for chemical fuel, and any such additional weight would have to be taken out of the payload.

In each of the cycles the attainable performance will be strongly affected by the temperature level at which the system can operate. As mentioned previously, a consistent set of temperature limits cannot be estimated at this time. A great amount of experimental work must be done before any temperature can be called a limit. Consequently, the results are presented first in terms of payload against temperature. Later, comparison curves are presented for selected temperatures. Similar plots showing the effects of variations in (1) airplane lift-drag ratio and (2) allowable dose rate are also included to illustrate just how sensitive the nuclear aircraft are to changes in these debatable parameters.

The matter of shield weight is still the biggest problem in any study of this kind, especially when unit shields are proposed. The shield weights used in this study were estimated by a very unsophisticated method, but the results coincided quite well with more detailed designs made by Pratt & Whitney and General Electric. Thus, in a sense, this method

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merely scaled these model shields to other reactor powers and core sizes. In this way trends can be shown that do not involve lengthy computation procedures.

The other assumptions and debatable numbers are of lesser uncertainty than the shield weight. An aircraft structure-to-gross-weight ratio of 0.3 was used for all systems, and a 20-percent thrust margin was reserved for maneuverability at the design point. The lift-drag ratio at Mach 2 was assumed to be 6.6, but the effect of variations in lift-drag ratio will be shown later. Engine weights and component efficiencies are equivalent for the three systems, with compressor pressure ratio being optimized. The sea-level static compression ratio was limited to the range from 2.5 to 15. Other component weights and configurations seem well enough understood that they will not be discussed here, but relative magnitudes are presented in table I.

The variations of payload with critical temperature are presented for the three systems in figure 3. Recall that the critical temperatures in both the air and lithium cycles are the reactor wall temperatures, whereas the critical temperature in the helium cycle is the maximum heat-exchanger wall temperature.

Figure 3(a) shows a plot of payload against effective reactor wall temperature for the direct air system. Curves for Mach numbers of 0.9 and 2.0 are given for an altitude of 35,000 feet and an airplane gross weight of 500,000 pounds. The actual limit is the maximum wall temperature, which is higher than the effective temperature by some unspecified amount. Keeping this difference small will be one of the most difficult design and development problems connected with this system.

The reactors for these direct-air-cycle aircraft are very large, because the airflows and powers are large. The shields are correspondingly heavy. That such would be the case could be anticipated from qualitative considerations. Thus a comparison which shows that the air cycle is not well suited to powering supersonic logistic carriers should not be surprising. However, it cannot be concluded that the air cycle is inferior for all flight conditions.

The curves show a large difference between performance at Mach 2 and at Mach 0.9. This difference is due to the great difference in airflow, a factor of 4.5.

The same type of plot is given for the lithium system in figure 3(b). Effective reactor wall temperature is again the abscissa; but, because of the excellent heat-transfer characteristics of lithium, not much difference between effective and maximum wall temperatures is expected. In contrast to the payloads for the direct air system, the values here are quite impressive. The lithium cycle is characterized by small reactors, low compression ratios, and high-effectiveness heat exchangers. The direct results are low shield and engine weights and turbine-inlet temperatures very near the lithium temperatures. Consequently, the number of engines can be changed greatly to make up for low specific impulse at low turbine-inlet temperature without a large effect on over-all airplane performance, because the engine weight is such a small part of the aircraft gross weight. Also, the low compressor pressure ratio moves the point at which performance decreases rapidly to relatively high Mach numbers or low reactor temperatures.

The payload for the helium system is plotted against maximum heat-exchanger wall temperature in figure 3(c). Payload values are roughly between those for the air and lithium systems. There is a large difference between performance at Mach 2 and at Mach 0.9 because the heat exchangers are a large part of the airplane gross weight (17% at Mach 2). As the airflow and power increase with Mach number, the frontal area and weight of the heat exchanger also increase, accounting for about half the reduction in payload.

The fairly rapid decrease in payload as the temperature decreases emphasizes the dominant role of the heat exchanger. Any improvement in heat-exchanger design would be very worthwhile in the helium system. A very compact counterflow tube-and-shell exchanger was used for these calculations for simplicity of analysis. Other geometries with extended surface may be expected to reduce the weight considerably, but the main problem is one of fabrication technique and design development.

An interesting result of the optimization of the helium system was that the best compromise between helium pumping power and reactor size was at a very high reactor pressure drop. At the Mach 2 and 1800° F point, the helium compressor power is 15 percent of the air compressor power. This illustrates the importance of shield weight even at small reactor diameters. The result of the power extraction is a relatively low specific impulse and a greater required airflow for the specified thrust.

At this point it becomes necessary to compare the three cycles at certain selected temperatures. The problem is in the proper selection of reference temperatures. Performance bands are plotted over arbitrarily chosen 400° spreads in figure 4. Payload is plotted against flight Mach number for the direct air cycle in figure 4(a). The upper boundary of the shaded band corresponds to an effective reactor wall temperature of 2200° F; the lower boundary corresponds to 1800° F. General Electric has already tested fuel elements at average temperatures around 1800° F, and higher temperatures are certainly to be expected. With respect to logistic carriers, direct air cycle would require a temperature of 2200° F or more to exhibit good payload capability at supersonic speeds.

The same type of band for the lithium system is superimposed on the air cycle plot in figure 4(b). The range of effective reactor wall temperature for the lithium cycle is from 1700° to 1300° F. If bimetal heat exchangers of good efficiency can be developed, performance in the upper part of the band is possible. If stainless steel or nickel-base alloys must be used, however, even the lower limit may be too high. Still, the lithium system shows very impressive performance over the entire range of temperature and Mach number illustrated.

In figure 4(c) the helium-system performance band is superimposed on the same plot. The range of maximum heat-exchanger wall temperature is from 2000° to 1600° F. The helium-system temperature level was chosen lower than the direct air temperature level because of the high pressure stresses the exchanger must withstand to contain the helium. For these temperatures the payload spread falls completely below the lithium band and only overlaps the direct-air-cycle curve slightly. If the helium-system heat exchanger could be substantially improved, the gap between the helium and lithium systems would be reduced considerably.

Figure 5 shows the variation of payload with altitude for the three cycles at a flight Mach number of 2 and the same temperature spreads as in figure 4. The gross weight is maintained at 500,000 pounds so that the principal effect of altitude is to change the weight of air-handling components. The result is a rapid decrease in payload with increasing altitude for the direct air and helium systems because of their high shield and heat-exchanger weights, respectively. The lithium-system performance deteriorates less rapidly with increasing altitude, because its engines and heat exchangers are a smaller fraction of the gross weight.

Since there is so much doubt regarding the accuracy of all unit shield weight estimates and since the values recommended for allowable dose rate are constantly being changed and debated, the variation of payload with dose rate has been plotted in figure 6. Because the shield is shaped, the dose rate is specified at 50 feet in the proper direction from the reactor. Curves for the three systems are all included on this plot, and again a band is plotted in preference to a single curve. The band width represents a spread of ±20 percent in shield weight. The middle line of each band corresponds to the midpoint of the temperature spread previously considered, and the design point for all three systems is Mach 2 and 35,000 feet. The shield weights used in the previously reported calculations were based on 0.5 rem per hour.

The slope of the bands indicates that there is little penalty associated with dose-rate variation for the lithium or helium systems. The direct air system, having a large shield, is more sensitive to changes in shield thickness. The widths of the bands reflect the same situation. In the two systems having small reactors, a 20-percent variation amounts to much less than 20 percent of the direct-air-system shield weight.

The effect of a variation in airplane lift-drag ratio is shown by figure 7. Recall that the lift-drag ratio used at Mach 2 was 6.6. As the lift-drag ratio is changed, a gross weight of 500,000 pounds is maintained, so that the lift is constant and the air-handling components must be increased or decreased to compensate for the change in drag. As in previous comparisons, the direct air and helium systems, having relatively large powerplant weights, are more drastically affected by changes in lift-drag ratio than the lithium system is.

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To illustrate the relative magnitudes of the component weights more clearly, weight breakdowns for the three systems are presented in table I. The same temperature spreads as used previously are shown for Mach 2 at 35,000 feet. The point made previously regarding the enormous shield weights in the direct air system is shown here. These reactors are 8 and 6 feet in diameter, and the direct air system is at a disadvantage with such high airflows. Use of two reactors would make the reactor diameter much more reasonable, but the total shield weight would be even higher than for the one large reactor. Note that, because of the high shield weight, the optimum compressor pressure ratios are 8 and 10 at design point.

The next two columns in table I are for the lithium system at 1300° and 1700° F effective reactor wall temperatures. The shield weights are low because of the small reactors (about 2.3 ft in diameter). The engine weights are also small because of the low compression ratio. Note also the relatively low heat-exchanger weight for comparison with the helium system.

The last two columns are for the helium system at 1600° and 2000° F maximum heat-exchanger wall temperature. The heat-exchanger weights are very much greater than the corresponding weights for the lithium system. The helium-to-air heat-exchanger weights could probably be reduced, perhaps by as much as 40 percent, with a corresponding reduction in the payload gap between the two systems. The helium-system reactors and shields weigh only slightly more than those for the lithium system, and the reactor cores are about 2.5 feet in diameter. Because of the high heat-exchanger weight, the optimum compression ratio for the helium system is about 3.8. Note again the payload values, increasing from air to helium to lithium for these temperature ranges.

No definite conclusions should be drawn from this study. Rather, some gross effects have been illustrated which point up the different characteristics of these three nuclear-propulsion systems. Some of the principal unknowns have also been emphasized, particularly with regard to the temperatures at which the systems can be operated. If lithium can be contained at temperatures in the range used for illustration, the lithium system shows very good performance in high-speed, medium-altitude aircraft, the helium system running a close enough second to become attractive if the heat-exchanger weight can be reduced or if the liquid-metal temperature limit should be lower than shown herein.

WEIGHT BREAKDOWNS

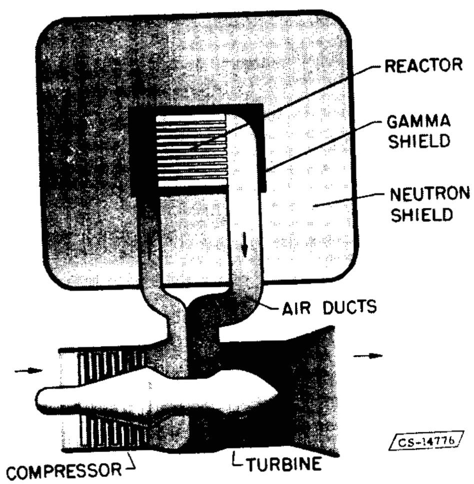
GROSS WT = 500,000 LB, MACH NO. = 2, ALTITUDE = 35,000 FT

SYSTEM	DIRECT AIR	LITHIUM	HELIUM
EFF. REACTOR WALL TEMP, °F	1800 - 2200	1300 - 1700	
MAX HEAT EXCH WALL TEMP, °F			1600 - 2000
COMP PRESS RATIO	8 - 10	2.2	3.8
REACTOR DIAM, FT	8 - 6	2.4 - 2.2	2.6 - 2.4
WEIGHT, THOUS OF LB			
TURBOJETS	60 - 37	27 - 15	47 - 32
HEAT EXCH		58 - 34	105 - 74
COOLANT CIRC SYST		39 - 17	24 - 20
REACTOR + SHIELD	290 - 216	87 - 83	97 - 88
TOTAL POWERPLANT	350 - 253	211 - 145	273 - 214
PAYLOAD	0 - 97	149 - 201	77 - 136

CS-14777

Table I

DIRECT AIR SYSTEM



DIRECT AIR SYSTEM

DISADVANTAGES

- POOR FLUID
- LARGE REACTOR, HEAVY SHIELD
- FUEL ELEMENT OXIDATION

ADVANTAGES

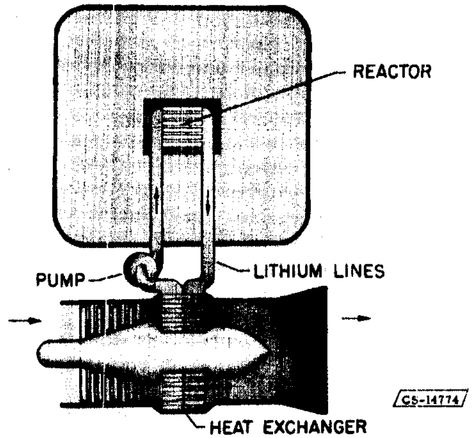
- SIMPLICITY
- NO INTERMEDIATE ΔT
- NO PUMPING WORK

CS-14773

Figure 1(a)

[redacted]

LITHIUM SYSTEM



LITHIUM SYSTEM

DISADVANTAGES

- ISOTOPE SEPARATION
- CORROSION, MASS TRANSFER
- HEAT EXCHANGER MATERIAL

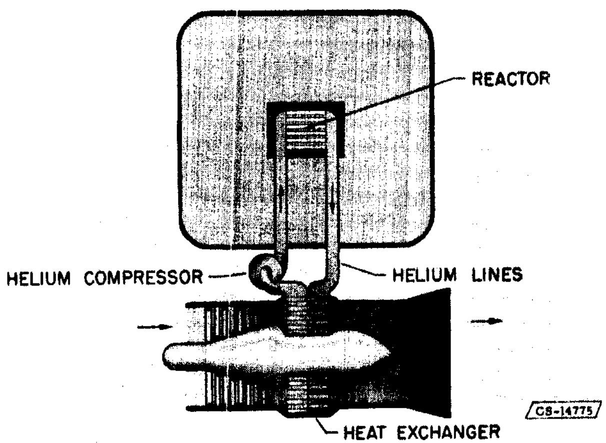
ADVANTAGES

- NO INTERMEDIATE HEAT EXCHANGER
- GOOD FLUID
- SMALL REACTOR, LIGHT SHIELD

CS-14771

Figure 1(b)

HELIUM SYSTEM



HELIUM SYSTEM

DISADVANTAGES

- HIGH PRESSURE
- HEAT EXCHANGER OXIDATION

ADVANTAGES

- SMALL REACTOR, LIGHT SHIELD
- INERT GAS
- NO INTERMEDIATE HEAT EXCHANGER

CS-14772

Figure 1(c)

[redacted]

RELATIVE PERFORMANCE OF REACTOR COOLANTS

REACTOR POWER = 700MW

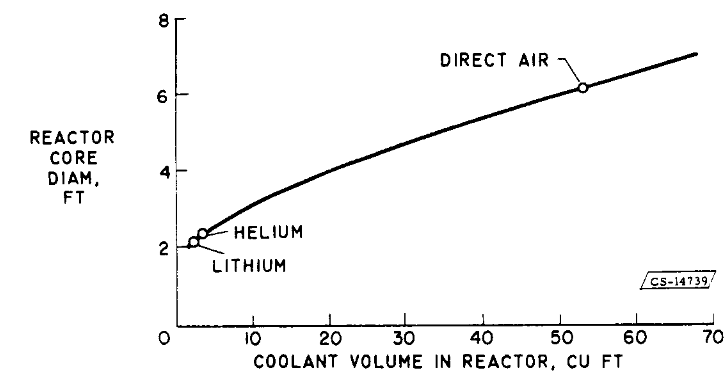


Figure 2

PAYLOAD VS TEMPERATURE
DIRECT AIR TURBOJET

GROSS WT = 500,000 LB, ALTITUDE = 35,000 FT

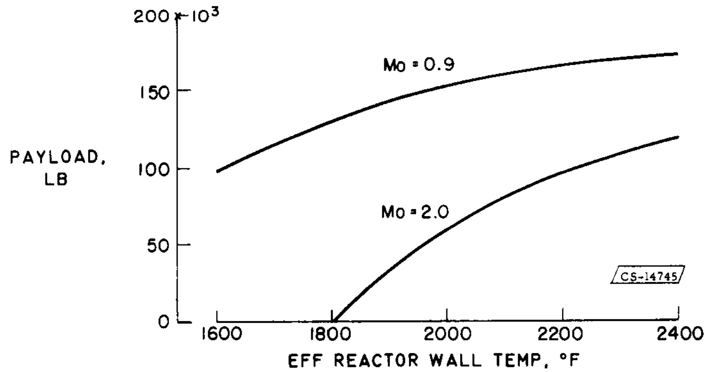


Figure 3(a)

PAYLOAD VS TEMPERATURE
LITHIUM TURBOJET

GROSS WT = 500,000 LB, ALTITUDE = 35,000 FT

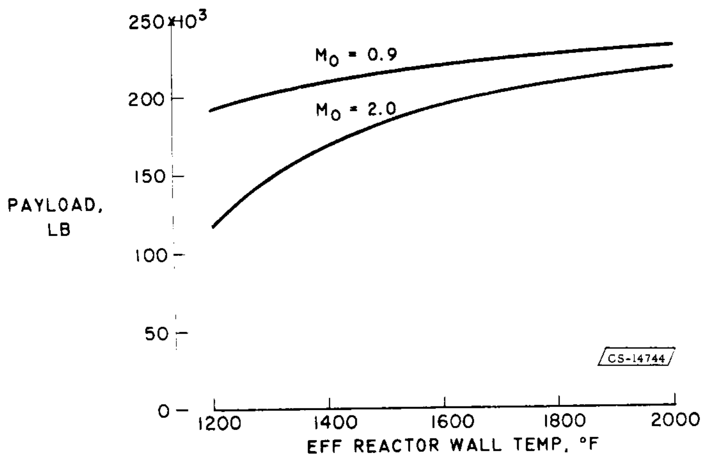


Figure 3(b)

PAYLOAD VS TEMPERATURE
HELIUM TURBOJET

GROSS WT = 500,000 LB, ALTITUDE = 35,000 FT

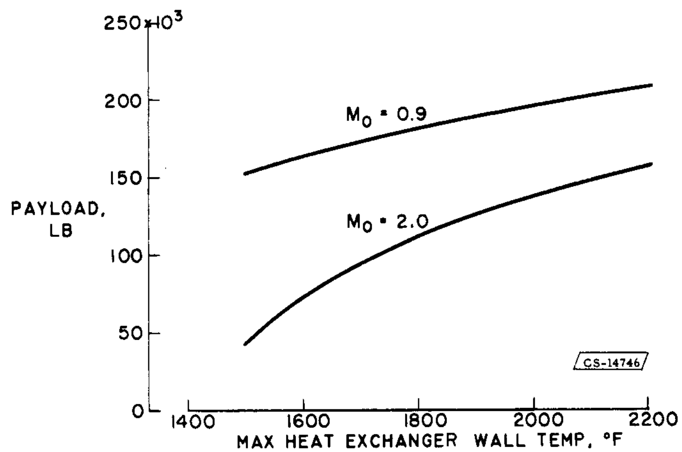


Figure 3(c)

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PAYLOAD VS FLIGHT MACH NUMBER

GROSS WT = 500,000 LB, ALTITUDE = 35,000 FT

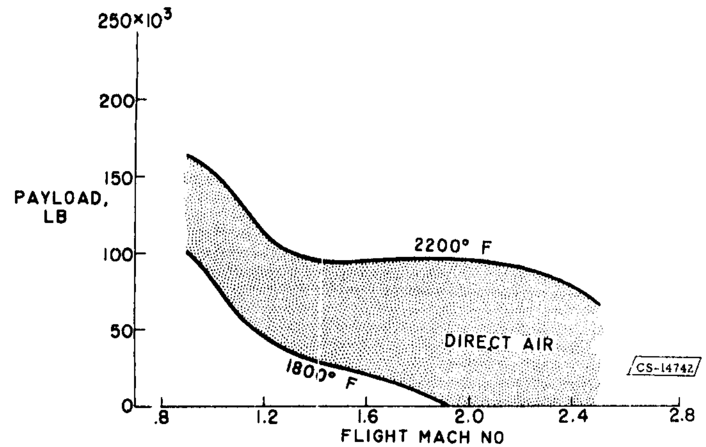

Figure 4(a)

PAYLOAD VS FLIGHT MACH NUMBER

GROSS WT = 500,000 LB, ALTITUDE = 35,000 FT

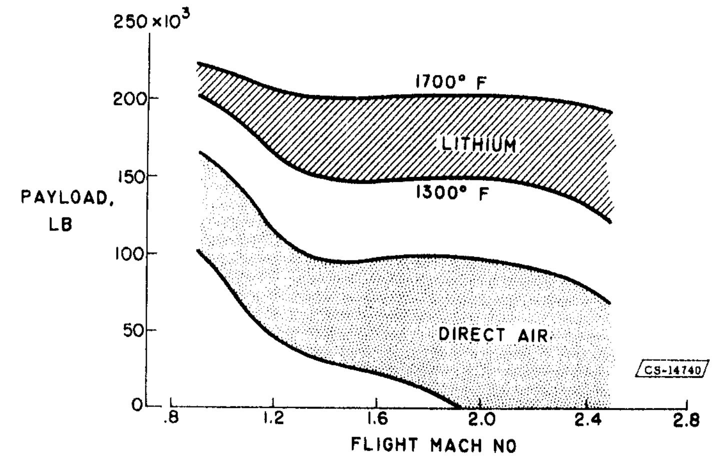

Figure 4(b)

PAYLOAD VS FLIGHT MACH NUMBER

GROSS WT = 500,000 LB, ALTITUDE = 35,000 FT

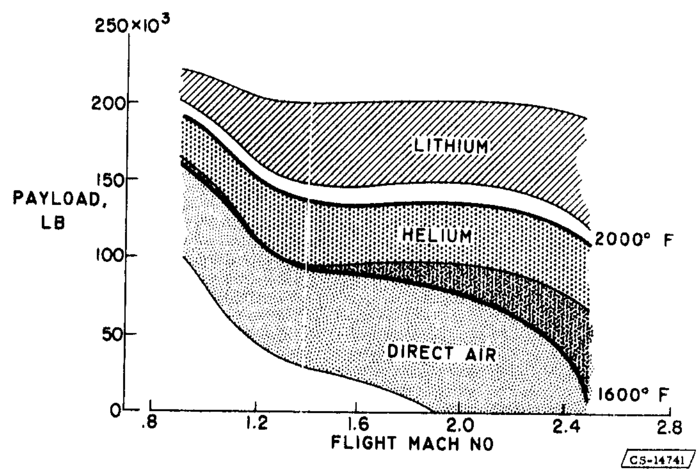

Figure 4(c)

PAYLOAD VS ALTITUDE

GROSS WT = 500,000 LB, MACH NO = 2.0

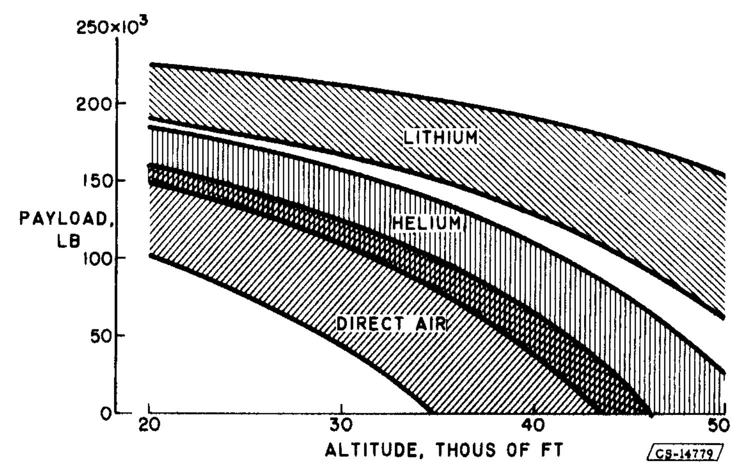

Figure 5

EFFECT OF VARIATIONS IN ALLOWABLE DOSE RATE AND SHIELD WEIGHT

MACH NO. = 2.0, ALTITUDE = 35,000 FT

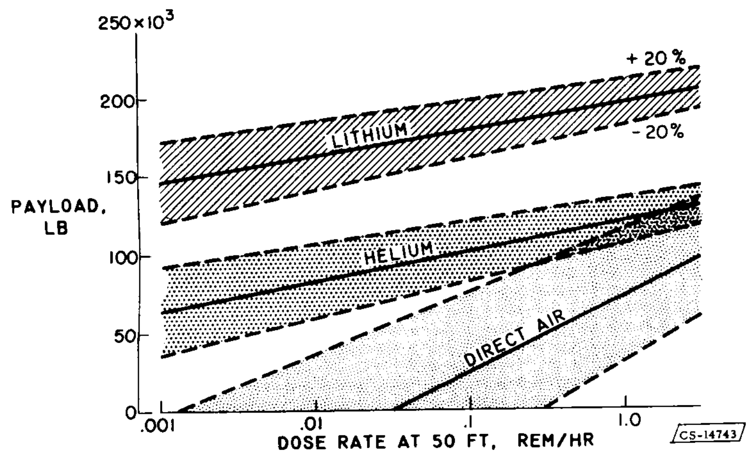

Figure 6

EFFECT OF VARIATION IN AIRPLANE LIFT/DRAG RATIO

GROSS WT = 500,000 LB, MACH NO = 2.0, ALTITUDE = 35,000 FT

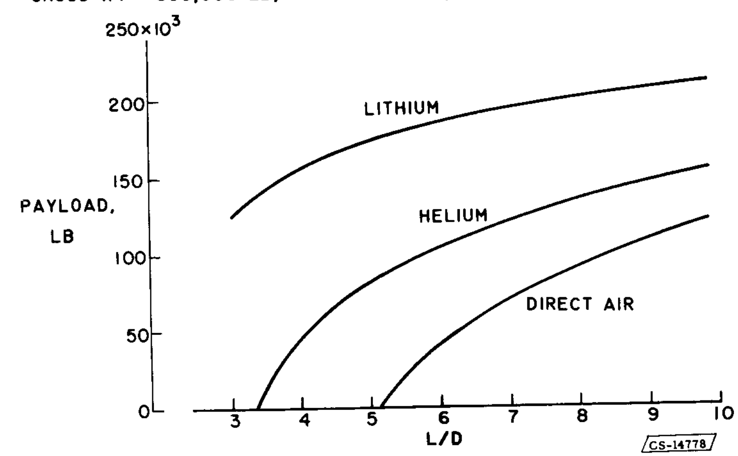

Figure 7

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[redacted] [redacted]

1. NUCLEAR LOGISTIC CARRIER

By Paul G. Johnson, James W. Miser,
and Roger L. Smith

[redacted]

[redacted]

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2. NUCLEAR ROCKETS

By Frank E. Rom, Eldon W. Sams, and Robert E. Hyland

The performance expected of nuclear rockets as determined by materials temperature limits is discussed in this paper. First, nuclear rocket powerplants for use in lifting payloads from the Earth's surface to an Earth satellite will be discussed. Reactors made of various materials will be compared to determine which materials hold the most promise. In addition, the feasibility of nuclear rockets for interplanetary flight will be discussed briefly.

In the previous papers on chemical rockets, the importance of high specific impulse was made clear. Figure 1 illustrates this need for an extended range of specific impulse. The weight breakdown of rockets required for two missions is plotted as a function of specific impulse. The ordinate represents weight expressed as a fraction of the total initial weight. The lower curve represents the Earth to Earth satellite mission. The top curve is the Earth satellite to Mars satellite and return mission. Both missions are single-stage missions. The area above each curve represents fuel weight, while the area below represents the remaining weight available for payload, structure, and engines.

Chemical rockets operating at a specific impulse of about 400 require fuel weights of 90 percent of the initial weight, leaving 10 percent for engines structure and payload. At a specific impulse of 1000 the fuel weight is reduced to 70 percent of the gross weight, thus tripling the weight allowed for engines, structure, and payload. Beyond a specific impulse of 3000, more than 70 percent of the initial weight can be engines, payload, and structure.

In order to obtain high specific impulses, it is desirable to use low-molecular-weight propellants operating at the highest possible temperature. Using fissioning uranium as a heat source, theoretically at least, permits practically unlimited temperatures and in addition permits the free choice of propellant.

Hydrogen, which is the lowest molecular weight element, can be heated by the fission source of energy. The resultant specific impulse of hydrogen as a function of temperature is shown in figure 2. The pressure of the hydrogen before expansion through the nozzle is 100 atmospheres, while the nozzle pressure ratio is considered to be infinite. The upper curve represents equilibrium expansion, while the lower curve represents frozen expansion. The actual specific impulse will be somewhere between the two curves, depending on the amount of recombination that takes place in the expansion process.

At a temperature of about 6000° F, which is approximately the limit imposed by materials, the specific impulse is about 1000. Beyond this temperature the reactors must be gaseous, and extraordinary methods are required for cooling the walls containing the gas; or else other methods for gas containment, such as by magnetic methods, are necessary.

Figure 3 shows schematically how a nuclear-powered rocket might look. The payload and guidance equipment are located in the nose. A large propellant tank contains the hydrogen in liquid form. A pump pressurizes the hydrogen and circulates it through the walls and other parts of the motor that require cooling. The hydrogen then is heated in a nuclear reactor and expanded through a nozzle to produce thrust.

The first case considered is that in which the hydrogen is heated by contact with solid materials containing fissioning uranium. Figure 4 shows a schematic drawing of such a system. The heart of the system is a nuclear reactor, which contains uranium in some solid form. The core, composed of a moderating material, is pierced with passages to permit the hydrogen to flow through and be heated. A neutron reflector with coolant passages is indicated around the sides of the core. A pressure shell surrounds the core and reflector. Thermal shielding is provided inside the pressure shell to reduce the gamma heating in the walls. Further gamma shielding is provided outside the shell in the direction of the pump and propellant tank to minimize heating in the pump and in the propellant. This shield also protects the payload, guidance equipment, and human beings from direct radiation. The pump to the right pressurizes the hydrogen and circulates it through the nozzle walls, the reflector, and along the thermal shield and pressure shell walls for cooling purposes. The hydrogen flows through the reactor where it is heated. The hot hydrogen then expands through the nozzle.

The key to obtaining high specific impulse is the use of the best possible high-temperature materials. Figure 5 shows approximate maximum operating temperatures for various materials that might be used in the reactor core. The first group of three materials represents moderating materials. Beryllium could be operated at 1700° F, beryllium oxide at 3300° F, and graphite at 5000° F. The reactor core could be made of these materials with uranium in some suitable compound dispersed throughout. The heat of the fissioning uranium would thus be generated directly within the moderator. Holes piercing the moderator would heat the hydrogen flowing through them. Beryllium and beryllium oxide are better moderators than graphite and should yield smaller reactors for a given propellant flow. Graphite, however, would produce higher temperatures. The relative importance of high temperature and small reactor size is discussed later.

On the right in figure 5 are listed structural materials that might be used as fuel elements if it is desired to contain uranium in metallic materials instead of dispersing it in the moderator. The nickel-base materials, which are fairly well developed, can be expected to operate at temperatures up to 2000° F. Molybdenum-base alloys, which have had little development work thus far, may be expected to reach temperatures approaching 4000° F. The tungsten-base alloys, about which very little is known at present, may some day reach 5500° F. The highest melting materials known, hafnium carbide and tantalum carbide, may someday provide operating temperatures of around 6000° F.

In the subsequent discussion the performance of graphite and beryllium oxide reactors with uranium dispersed in the moderator itself is examined first. Then reactors that use the better moderating materials as moderators and contain the uranium in fuel elements made of molybdenum and tungsten are considered. The mission that will be used as the basis of the comparison is the carrying of large payloads from the Earth's surface to a satellite orbit about the Earth.

The discussion of the performance that can be obtained from a given reactor will be based on the reactor shown in figure 6. The core is composed of uranium-impregnated graphite with holes piercing it for passage of the hydrogen. The core diameter is 3.5 feet and length is 2.8 feet. The flow area represents 30 percent of the frontal area of the core. The reflector chosen is a 6 inch-thick beryllium reflector. This reflector material and thickness result in about the minimum core-plus-reflector weight for the given hydrogen flow area desired. The uranium investment is about 77 pounds.

In the operation of this reactor, the operating temperature level, the hydrogen flow velocity, and the pressure level may be selected. The temperature is determined by materials limitations. The best hydrogen flow velocity, which is a result of performance calculations, is that value which gives very near choking conditions at the reactor exit. In all the subsequent calculations the best hydrogen velocity will be used.

The choice of the best pressure level is illustrated in figure 7. The effect of hydrogen pressure on powerplant weight and thrust per engine weight for a maximum surface temperature of 5000° F is shown. The powerplant weight includes the reactor core and reflector shown in figure 6, and also the pressure shell, nozzle, turbopump unit, and shielding necessary to reduce heat generation in the pressure shell and in the propellant. The shielding also affords protection from direct radiation to the payload, guidance equipment, and human cargo. The increase in powerplant weight with pressure is due to the increase in pressure shell, nozzle, and turbopump weight with pressure level. The thrust per powerplant weight ratio increases with pressure level in spite of the increased powerplant weight because of the overriding effect of the thrust increase.

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At a pressure level of 1200 pounds per square inch, the thrust to powerplant weight ratio is 30, and the powerplant weight is about 17,000 pounds.

Figure 8(a) shows the performance that can be expected of this same rocket motor as a function of pressure. The mission is to establish a satellite about the Earth with a single-stage vehicle. The thrust-to-gross-weight ratio chosen is 2.0, which is about best as determined by a series of calculations. The payload and gross weight both increase with increasing pressure, reflecting the increase in thrust due to the pressure increase. At a pressure of 1200 pounds per square inch with an initial gross weight of 260,000 pounds, it is possible to carry a payload of 20,000 pounds to a satellite.

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In the next case, the gross weight is held constant at 300,000 pounds. The thrust and hydrogen flow required are constant, so that increasing pressure reduces the required reactor size. The reactor diameter and payload are plotted in figure 8(b) as a function of pressure for the Earth satellite mission. The reactor diameter decreases as shown, from about 4.1 feet to about 3.2 feet when the pressure is increased from 800 to 2000 pounds per square inch. The payload increases from about 20,000 to 25,000 at 1300 or 1400 pounds per square inch, and then decreases slightly beyond this pressure. An optimum pressure is indicated, but the curve is quite flat. The reason the payload curve shows an optimum is as follows: At first, the reduction in core size reduces the powerplant weight, giving a higher payload. As pressure increases further, the increase in pressure shell, nozzle, and turbopump weight is more important than the reduction in core weight, and the payload weight decreases.

The operating pressure to be selected, then, does not come from calculations such as these. The pressure is determined by practical limitations such as (1) the problem of pumping cryogenic fluids to very high pressures and (2) the problem of designing cooled pressure shells with internal gamma heat generation.

The discussion thus far has been based on the use of uranium-impregnated graphite as the reactor core material. It might be suggested that beryllium oxide should be used in place of graphite, since it is a much better moderating material than graphite. The use of beryllium oxide would reduce the required core size for a given hydrogen flow. However, since the operating temperature is much lower for beryllium oxide, the specific impulse would be less.

The powerplant weight and thrust per powerplant weight for graphite and beryllium oxide reactors with dispersed uranium are plotted as a function of hydrogen flow rate in figure 9 for a pressure of 1200 pounds per square inch. The beryllium oxide reactors operate with a maximum surface temperature of 3300° F with a specific impulse of 645 seconds. The

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graphite reactors operate with a maximum surface temperature of 5000° F and with a specific impulse of 816 seconds. The graphite reactors are about 50 percent heavier than the beryllium oxide reactors because of the superior nuclear characteristics of beryllium oxide. The fact that the specific impulse is lower for the beryllium oxide does not overcome the lower weight advantage, as shown by the higher value of thrust per powerplant weight for beryllium oxide. On the basis of thrust per powerplant weight ratio, beryllium oxide would appear to be the better propulsion system.

In considering a rocket vehicle, the specific impulse must also be taken into account. Table I shows the performance of Earth to Earth satellite rockets using the dispersed-uranium graphite and beryllium oxide reactors of figure 9. Reactor sizes were chosen to obtain the thrust required for a 300,000-pound-initial-weight single-stage rocket. The maximum surface temperature and specific impulses are again noted for the beryllium oxide and graphite reactors. The beryllium oxide powerplant has a thrust to powerplant weight ratio about 30 percent greater than the graphite powerplant, but has a 20-percent-lower specific impulse. The net effect is that the payload of the beryllium oxide system is about 60 percent less than the payload for the graphite reactor. Thus, it may be concluded that, if reactors in which the uranium is dispersed throughout the moderator are to be used, graphite is the better material to use.

The use of beryllium oxide results in very substantial powerplant weight savings. In order to take advantage of this, the uranium must be removed from the moderator and placed in high-temperature materials fabricated into fuel elements. The high-temperature fuel elements then heat the hydrogen. The moderator must be cooled in this case. A schematic picture of one such system is shown in figure 10. The rocket motor pictured is similar to the previous one in all respects except that the core arrangement is different. The uranium is contained in high-temperature materials such as molybdenum or tungsten fabricated into flat plates, concentric sheets, or tube bundles. The elements are located in holes in the moderator. The hydrogen first passes from left to right in the annular gap between the hole and the fuel element. During this passage the heat generated in the moderator is picked up. The flow is then reversed and passes through the fuel element, which heats the hydrogen to the desired operating temperature. Because of the two coolant passes required, the flow area required in the reactor is about 20 percent larger than the once-through flow area, assuming that 5 percent of the heat produced is generated in the moderator. This penalty is included in all subsequent calculations for cooled-moderator reactors.

Figure 11 shows the performance expected with beryllium as the moderator and tungsten or molybdenum as the fuel-element material. Beryllium was chosen in place of beryllium oxide because its moderating ability is about the same as beryllium oxide but its density is lower, resulting in a lighter reactor. The powerplant weight and thrust to powerplant weight ratio are plotted as functions of hydrogen flow. The pressure level is again 1200 pounds per square inch. The dispersed-uranium graphite reactor is shown for reference. The tungsten fuel elements are assumed to operate at a maximum surface temperature of 5500° F, which produces a specific impulse slightly greater than that for the graphite reactor. The molybdenum fuel element operates at a maximum temperature of 4000° F and produces a specific impulse of 715 seconds.

The powerplant weights for the beryllium reactors are about 30 to 50 percent lower than the graphite reactor weights. The tungsten reactor is slightly heavier than the molybdenum reactor, because the hydrogen requires a larger flow area with tungsten, since the hydrogen is at a higher temperature. The thrust per powerplant weight for the beryllium reactors is about 40 percent higher than for the graphite. The value for the tungsten reactor is higher than for the molybdenum reactor, because the higher specific impulse more than offsets the slightly greater weight of the tungsten reactor.

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Thus, using a cooled beryllium moderator increases the thrust per powerplant weight of the rocket engine. Using tungsten for the fuel-element material increases specific impulse. Both of these effects should give better rocket performance.

Table II shows the performance expected of tungsten-beryllium and molybdenum-beryllium reactors compared with that of the dispersed-uranium graphite system. The comparison again is made for the Earth satellite mission with a 300,000-pound-initial-weight single-stage rocket. The thrust to powerplant weight ratio is about 25 percent higher for both beryllium-moderator systems than for the graphite system. This, coupled with higher specific impulse of the tungsten system, increases the payload from 28,500 pounds for the graphite system to 38,000 pounds for the tungsten-beryllium system. The molybdenum system has a payload about 25 percent less than the graphite system.

If it is desired to carry men to an Earth satellite, additional shielding would be required for protection against scattered radiation in passing through the Earth's atmosphere. Shielding against direct radiation is already provided for in the powerplant assembly weight. Approximately 35,000 pounds of additional shielding and equipment is required for a load of four men. This mission could be accomplished with the tungsten-beryllium reactor for the gross weight of 300,000 pounds. The graphite reactor would require a somewhat larger gross weight (about 350,000 lb).

It appears that ultimate nuclear rocket performance may come from the use of tungsten-base fuel elements in conjunction with good moderator materials that are cooled. The gains indicated are sufficient to warrant a closer look at the problems of such a reactor system for nuclear rocket propulsion.

There are additional advantages in using metallic fuel elements. Should it be desirable to design nuclear rocket systems for reuse, for example as a ferry from the Earth to an Earth satellite and return, it would be necessary to contain the fission products and uranium. Metallic fuel elements can contain these materials better than ceramic or graphite elements and therefore should be of greater interest. In addition, the metallic fuel elements are advantageous in that they are not attacked by hydrogen and so would not require protective coatings as is necessary with graphite. The disadvantage, of course, is the added complication of a cooled moderator.

The technology of molybdenum and tungsten is in its infancy, and a great deal of research is necessary to develop satisfactory alloys for operation at the temperatures indicated and also to learn how to fabricate and form these materials into reliable fuel elements.

Thus far, only an Earth to Earth satellite mission has been discussed. Now the possibility of using nuclear rockets for interplanetary flight will be considered. As will be pointed out in the next paper, high thrusts are not required to achieve flights to the moon or to Mars if the vehicle starts from an Earth satellite. High specific impulses are important, however.

It is possible to increase the specific impulse of nuclear rockets by operating them at lower pressure levels, because hydrogen can be dissociated into hydrogen atoms more readily at lower pressures. This is shown in figure 12, where specific impulse is plotted as a function of hydrogen temperature and pressure. In the range of 5000° F or higher, large increases in specific impulse are possible by reducing the pressure from 100 to about 1 atmosphere. For example, at a temperature of 5000° F the specific impulse can be increased from about 900 seconds at 100 atmospheres to about 1100 seconds at 1 atmosphere and to about 1400 seconds at 0.01 atmosphere.

To illustrate the use of low-pressure nuclear rockets for interplanetary flight, a mission to Mars from an Earth satellite and return to the Earth satellite will be considered. The mission consists in sending an eight-man exploring party to Mars with equipment for surface exploration of Mars. This mission will be described in greater detail in the next paper.

The reactor will be the same type as used for the satellite mission. Two beryllium-tungsten reactors that normally produce 800,000 pounds of thrust each at a pressure level of 1200 pounds per square inch will be operated at pressures of about 2 or 3 atmospheres. The weight of two of these powerplants would be about 40,000 pounds.

[redacted]

The performance expected for the Mars trip for specific impulses of 1000 and 1200 seconds is shown in table III. The required initial gross weights are 620,000 and 520,000 pounds, assuming that the required velocity changes occur instantaneously. Gas temperatures of 5000° and 5700° F with reactor pressures of 3.3 and 2.3 atmospheres, respectively, produce thrust equal to 10 percent of the gross weight, which then would give an acceleration of 0.1g. Taking into account the fact that the thrust is not applied instantaneously would increase the gross weights somewhat. The reactor powers are 1200 and 1700 megawatts, respectively.

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This Mars mission starts from an Earth satellite. In order to assemble the vehicle for the Mars journey, about 600,000 pounds of fuel and equipment must be placed in this satellite orbit by rockets from the Earth's surface. Nuclear rockets such as the 300,000-pound-gross-weight beryllium-tungsten rocket described earlier can be used. Each of these can carry a payload of 38,000 pounds. It would therefore take 15 trips to carry the fuel and equipment. An additional two trips would be necessary to place the eight men with shielding into the orbit.

Temperatures of 5000° and 5700° F are within reason with the use of tungsten or the carbides of hafnium and tantalum. It is easier to obtain these temperatures with low-pressure operation than at the high-pressure condition because of two effects. First of all, the reactor is operating at a lower power level, about 1/8 or so of the power at full pressure. This means that the temperature difference within the fuel element will be lower, so that the surface can be operated at a higher temperature without danger of melting the center. In addition, hot spots that develop owing to imperfections in the reactor construction will have a greater heat-removal rate, since the hydrogen will be dissociating at a greater rate at the hot spot. Dissociation, then, will tend to make temperatures more uniform throughout the reactor and thus increase the chances of obtaining higher gas temperatures.

It seems, then, that interplanetary travel with nuclear rockets limited to temperatures imposed by materials gives reasonable performance as indicated by gross weights in the range of 600,000 pounds for the Mars trip.

Further increases in performance can be obtained by going to higher specific impulses. The temperatures required, however, are beyond materials capabilities. In this temperature range and higher, the fissioning uranium must be in the gaseous phase and the heat must be transferred directly to the propellant. Several organizations are investigating methods of obtaining these ends. The NACA is investigating the possibility of heating hydrogen directly with fissioning uranium in gaseous form in the so-called cavity-type reactors. The chief problem is determining methods for preventing the uranium from escaping with the hydrogen. The use of centrifugal fields and magnetic fields for uranium retention is being sutdied.

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Another problem is the determination of the heat transferred by radiation from the gas mixture to the walls. In addition, the calculations of the criticality of cavity reactors is receiving some attention so that the required uranium concentration and power generation distributions can be determined.

In conclusion, nuclear rockets can be expected to carry payloads of about 10 to 13 percent of the initial gross weight to an Earth satellite. The nuclear rocket shows promise of Earth satellite to Mars and return interplanetary flights with initial gross weight within reason. This flight can be accomplished by reducing operating pressure so that dissociation effects can result in high specific impulses.

The use of nuclear energy as a heat source in heat-transfer rockets as presently conceived does not even begin to use the ultimate potential of the fission process. New ideas and concepts are required to utilize the full potential of nuclear energy for rocket propulsion.

ROCKET PERFORMANCE COMPARISON

DISPERSED URANIUM REACTORS, EARTH SATELLITE MISSION
PRESSURE, 1200 PSI, INITIAL GROSS WEIGHT, 300,000 LB

	BeO	GRAPHITE
MAX SURFACE TEMP, °F	3300	5000
SPECIFIC IMPULSE, SEC	645	816
THRUST TO P.P. WT. RATIO	38.5	30.0
PAYLOAD, LB	11,000	28,500
REACTOR POWER, M	9800	12,000

CS-14781

Table I

ROCKET PERFORMANCE COMPARISON

FUEL ELEMENT, COOLED MODERATOR TYPE,
EARTH SATELLITE MISSION; PRESSURE, 1200 PSI,
INITIAL GROSS WT, 300,000 LB

	Be-W	Be-Mo	GRAPHITE
MAX SURFACE TEMP, °F	5500	4000	5000
SPECIFIC IMPULSE, SEC	855	715	816
THRUST TO POWERPLANT WEIGHT RATIO	38.0	37.6	30.0
PAYLOAD, LB	38,000	21,000	28,500
REACTOR POWER, MW	12,700	10,800	12,000

CS-14749

Table II

LOW PRESSURE NUCLEAR ROCKET

FUEL ELEMENT, COOLED MODERATOR TYPE, MARS MISSION
ENGINE WT, 40,000 LB, THRUST TO INITIAL WT, 0.1

SPECIFIC IMPULSE, SEC	1000	1200
INITIAL GROSS WT, LB	620,000	520,000
THRUST, LB	62,000	52,000
TEMP, °F	5000	5700
PRESSURE, ATM	3.3	2.3
REACTOR POWER, MW	1200	1700

CS-14874

Table III

ROCKET WEIGHT BREAKDOWN

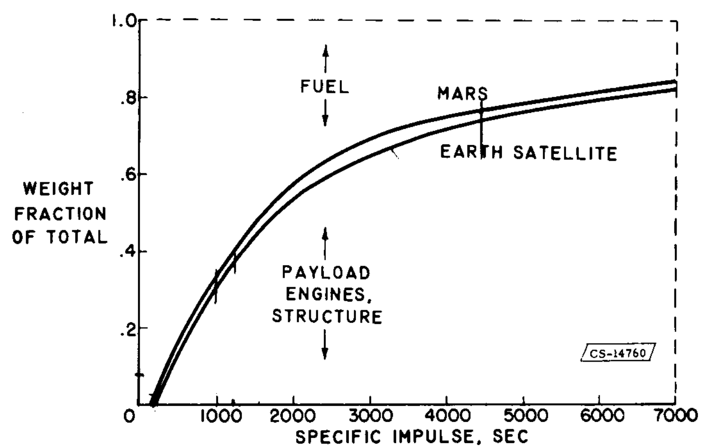

Figure 1

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SPECIFIC IMPULSE OF HYDROGEN

PRESSURE, 100 ATM

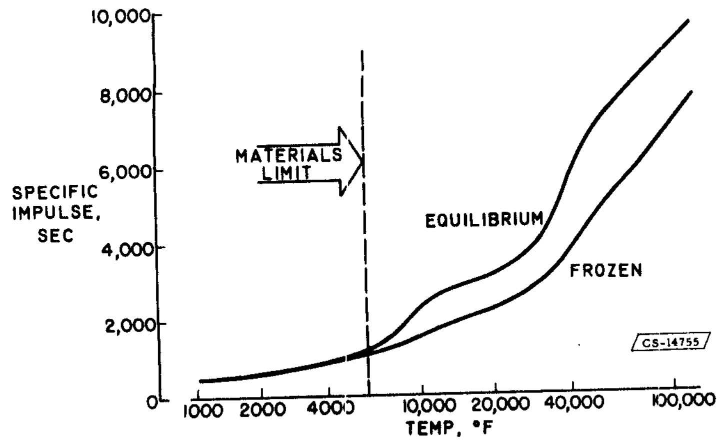


Figure 2

NUCLEAR ROCKET

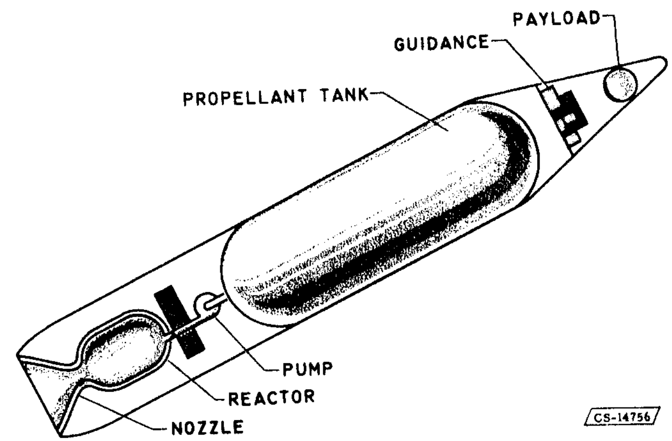


Figure 3

NUCLEAR ROCKET MOTOR

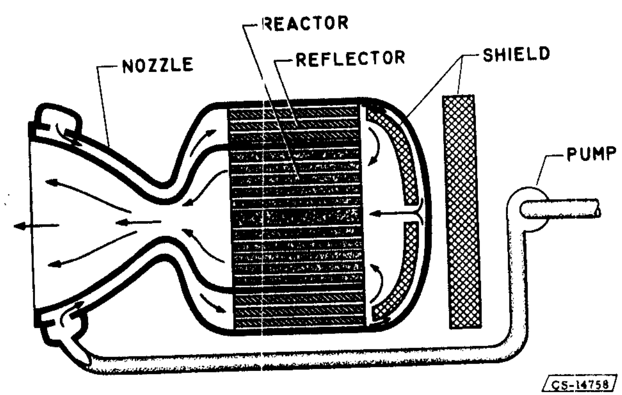


Figure 4

MAXIMUM TEMPERATURE FOR REACTOR MATERIALS

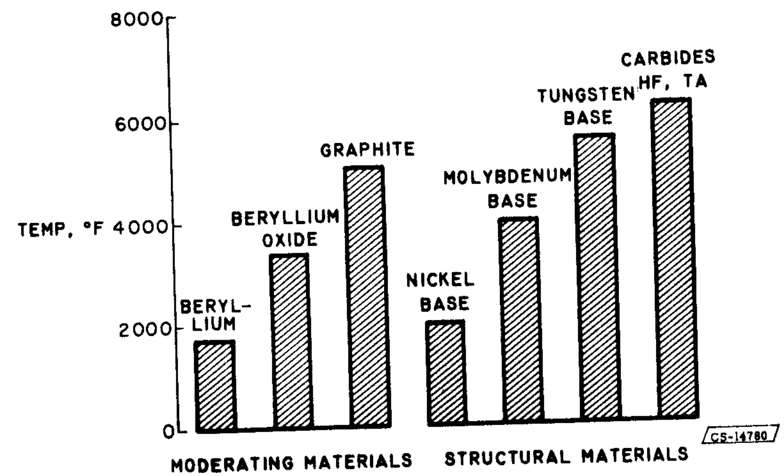
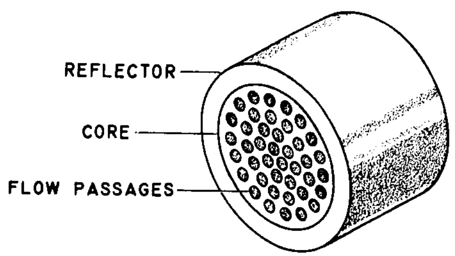


Figure 5

NUCLEAR ROCKET REACTOR

GRAPHITE, DISPERSED URANIUM



CORE DIAM., FT	3.5
CORE LENGTH, FT	2.8
CORE FREE-FLOW RATIO	0.3
REFLECTOR, Be, IN.	6
URANIUM INVESTMENT, LB	77

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Figure 6

EFFECT OF PRESSURE ON POWERPLANT PERFORMANCE

GRAPHITE, DISPERSED URANIUM
MAX SURFACE TEMP, 5000° F, D_R, 3.5', L_R, 2.8'

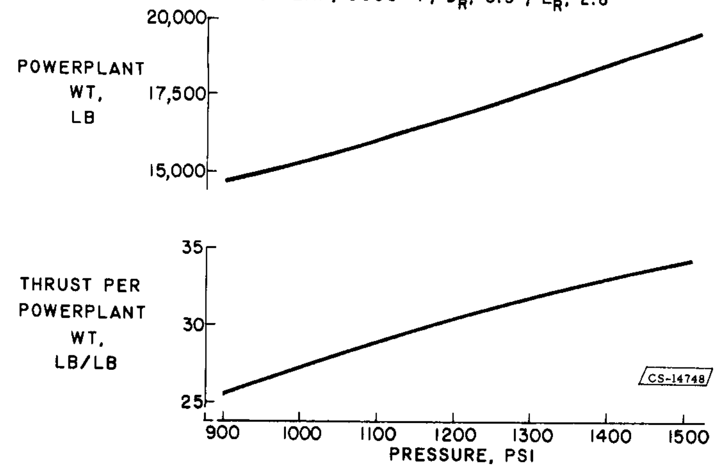

Figure 7

EFFECT OF PRESSURE ON ROCKET PERFORMANCE

GRAPHITE, DISPERSED URANIUM, EARTH SATELLITE MISSION
MAX SURFACE TEMP, 5000° F, D_R, 3.5', L_R, 2.8'

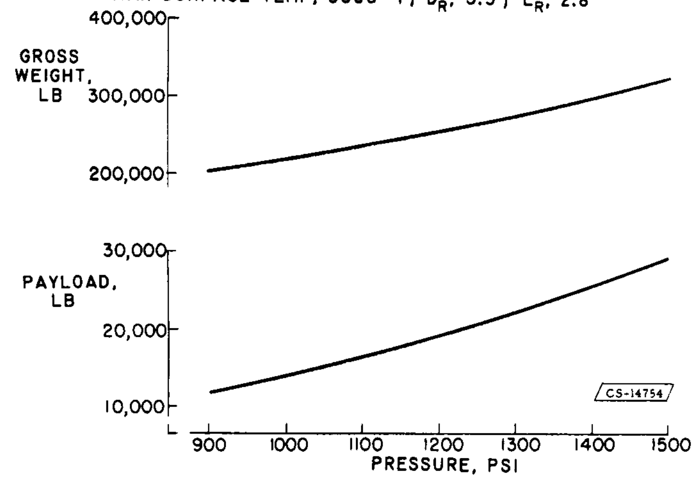

Figure 8(a)

EFFECT OF PRESSURE ON ROCKET PERFORMANCE

GRAPHITE, DISPERSED URANIUM, EARTH SATELLITE MISSION
MAX SURFACE TEMP, 5000° F, INITIAL GROSS WEIGHT, 300,000 LB

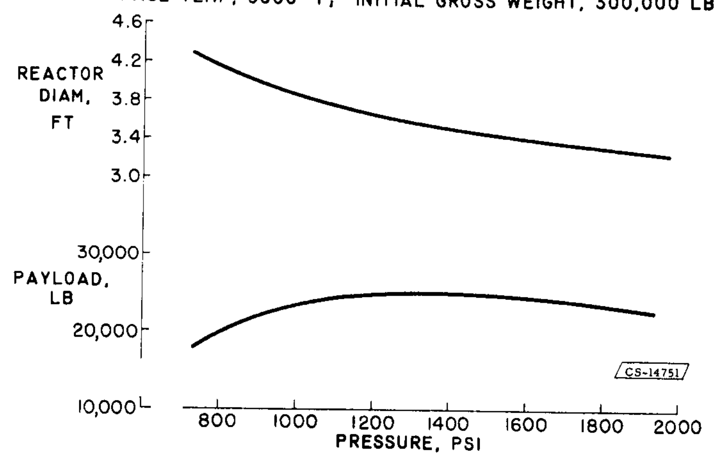

Figure 8(b)

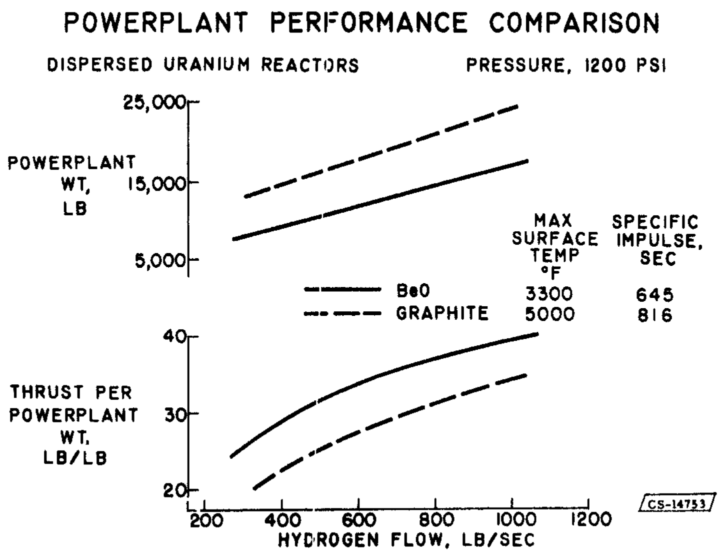

Figure 9

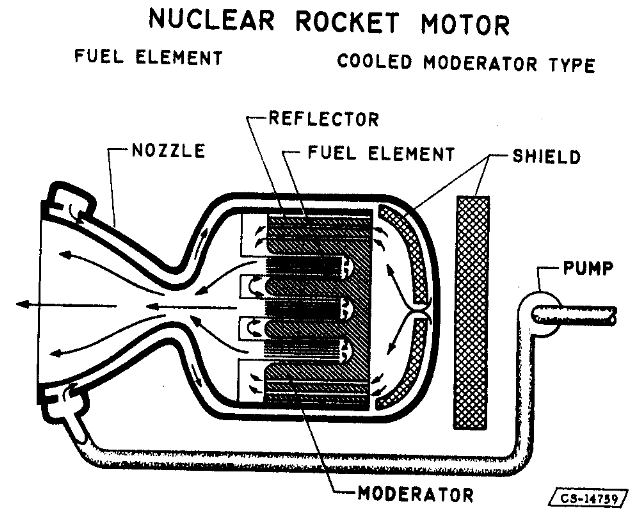

Figure 10

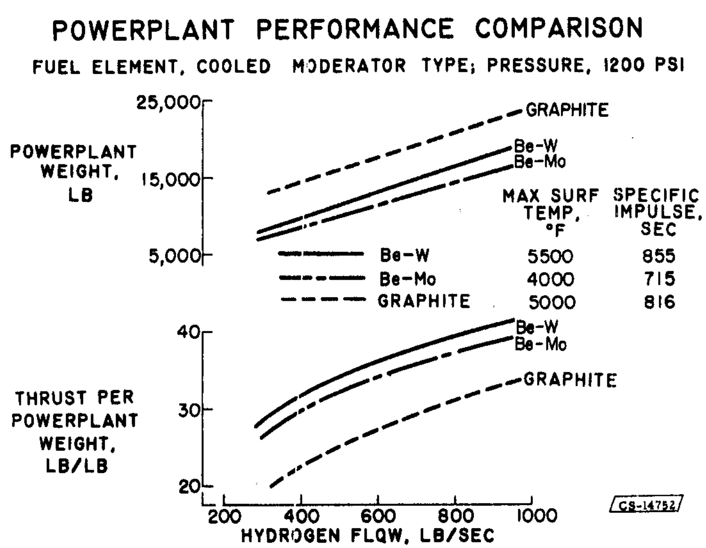

Figure 11

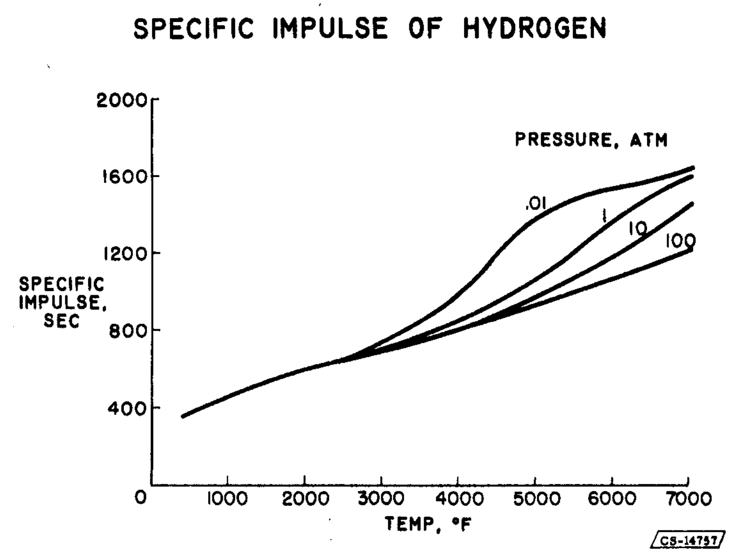

Figure 12

[redacted]

[redacted]

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3. SATELLITE AND SPACE PROPULSION SYSTEMS

By W. C. Moeckel, L. V. Baldwin, R. E. English,
B. Lubarsky, and S. H. Maslen

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3. SATELLITE AND SPACE PROPULSION SYSTEMS

By W. E. Moeckel, L. V. Baldwin, R. E. English,
B. Lubarsky, and S. H. Maslen

INTRODUCTION

Previous papers have described rocket systems capable of launching sizable payloads into satellite orbits. Propulsion systems that might be suitable for the next steps are discussed herein. Some of the uses for propulsion systems once satellites have been established are as follows:

(1) Increasing lifetime of low-altitude satellite

(2) Controlling and altering satellite orbits

(3) Lunar and interplanetary exploration

(4) Auxiliary electric power

Maintaining satellites in relatively low orbits, say of the order of 100 miles in altitude, may be desirable for observation of the Earth or as a missile-launching platform. At such altitude lifetime would be short unless a small, long-duration thrust is provided to overcome the drag.

Altering or controlling satellite orbits at higher altitudes may be desired, to correct perturbations or launching errors or to reorient the satellite orbit into a more favorable location for launching vehicles to other bodies in the solar system. Again, a very small but continuous thrust would be adequate unless a rapid change in orbit is desired.

Lunar and interplanetary expeditions could range in magnitude from a small instrumented one-way vehicle to a fully manned expedition capable of landing on and exploring another planet. Again, since the journey will start from an established satellite orbit, a small but long-duration thrust will suffice to move the vehicle out of its original orbit and eventually out of the Earth's gravitational field.

In addition to these propulsion applications, auxiliary electric powerplants will be needed aboard the vehicles to operate instruments and to control the environment in manned vehicles.

A variety of propulsion systems might be suitable for these purposes. The chemical and nuclear rockets, which have already been discussed, are capable of undertaking all of these missions. The recombination or solar ramjet might be used to sustain a satellite in an orbit at relatively low altitude. Various electric systems are possible for all the missions that have been mentioned.

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Most of this discussion will be devoted to various types of electric propulsion systems, chiefly because there are so many possibilities and because some of them look quite promising. The possibilities for electric propulsion systems are listed in the following table:

Basic energy sources	Electric power generators	Thrust generators
Chemicals	Chemical batteries	Electric-arc chambers
Radioisotopes	Radioisotope batteries	Ion accelerators
Solar radiation	Thermopiles	Plasma accelerators
Nuclear fission	Solar batteries	Photon accelerators
Nuclear fusion	Turboelectric generators	
	Induction from moving plasma	

These basic energy sources, electric power sources, and thrust generators can be combined in a variety of ways. Most of the feasible combinations will be discussed later; but, since there are so many possible systems, it is desirable to discuss their common characteristics first.

GENERAL CHARACTERISTICS OF ELECTRIC PROPULSION SYSTEMS

The two most significant common characteristics of electric propulsion systems are

(1) Higher specific impulse (lower propellant weight):

$$\text{Propellant weight} = \frac{(\text{Thrust})(\text{Propulsion time})}{\text{Specific impulse}} = \frac{F\tau}{I} \qquad (1)$$

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(2) Higher powerplant weight:

$$\frac{\text{Powerplant weight, lb}}{\text{Jet power, kw}} = \text{Specific weight, } \alpha \tag{2}$$

$$\text{Jet power} = \frac{FI}{45.8} \tag{3}$$

Hence,

$$\text{Powerplant + propellant weight} = F\left(\frac{\alpha I}{45.8} + \frac{\tau}{I}\right) \tag{4}$$

Electric propulsion systems can achieve higher specific impulse than chemical or nuclear rockets. Specific impulse is the velocity of the ejected particles divided by g; and ions, for example, can be accelerated to almost any desired velocity by sufficient voltage. Therefore, a given mission can be achieved with less propellant weight with an electric system than with chemical or nuclear rockets. The propellant weight is defined in equation (1). It might be assumed that the highest possible specific impulse is desired in order to reduce propellant consumption. That this is definitely not the case, however, may be seen when the second characteristic (i.e., the higher powerplant weight of electric systems) is considered.

As a figure of merit for the powerplant weight, the powerplant weight divided by the jet power is used (eq. (2)). This is somewhat different from the usual definition of specific weight, which is generally defined as the powerplant weight per unit thrust. For electric systems, however, the weight depends on the electric power produced; therefore, weight per unit power is a much more convenient definition. This specific weight is denoted by α (lb/kw). If the electric power were converted into jet power with 100-percent efficiency, α would be the same as the usual definition of specific weight for auxiliary electric powerplants (i.e., weight per unit of electric power produced). This α, then, takes into account the additional inefficiency of conversion of electric power into jet power.

The jet power is proportional to thrust times specific impulse (eq. (3)). This definition is convenient for the purposes of this paper, because it shows that, for a given required thrust, the jet power increases directly with the specific impulse. This means that, if there is a fixed weight per kilowatt, the total powerplant system will increase in weight as the specific impulse is increased. Adding the propellant weight and the powerplant weight equations gives equation (4), in which the powerplant term increases with specific impulse and the propellant term decreases. Thus, indefinitely high specific impulse is not desirable, because the powerplant weight could become too high even though the propellant weight decreased almost to zero. In fact, an optimum specific impulse would be expected that minimized the total powerplant plus propellant weight.

With equation (4) for powerplant plus propellant weight, the following equation can be written for the payload weight ratio:

$$\frac{W_{pay}}{W_0} = 0.95 - \frac{F}{W_0}\left(\frac{\alpha I}{45.8} + \frac{\tau}{I}\right) \tag{5}$$

The factor 0.95 is due to an allowance of 5 percent for structure and miscellaneous weight. The exact value of this factor is not as important for electric systems as for chemical or nuclear rockets, because the powerplant weighs much more than the structure. Since the powerplant must be carried throughout the mission, not much benefit can be expected from staging operations.

If the specific powerplant weight α and the propulsion time τ are temporarily assumed to be independent of specific impulse, the following equation for optimum specific impulse results:

$$I_{opt} = 1990\sqrt{\frac{\tau_{days}}{\alpha}} \tag{6}$$

This value minimizes the powerplant plus propellant weight and therefore maximizes the payload weight ratio. With this optimum specific impulse, the following equation for the maximum payload ratio is obtained for the electric propulsion systems:

$$\left(\frac{W_{pay}}{W_0}\right)_{max} = 0.95 - 87\,\frac{F}{W_0}\sqrt{\alpha\,\tau_{days}} \tag{7}$$

It is a function only of initial thrust to total weight ratio, the specific powerplant weight α, and the required propulsion time τ. An interesting point is that, for this optimum specific impulse, the propellant weight is equal to the powerplant weight.

The assumption that α and τ are independent of specific impulse is not generally valid, but it is shown later that other assumptions and more accurate analyses give about the same minimum powerplant plus propellant weight.

In order to determine the payload ratio for various missions, three parameters must be determined: the initial thrust-weight ratio, the propulsion times required to accomplish the mission, and the specific weight

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of the powerplant. The specific weights attainable with various propulsion systems will be considered later, but first the required values of thrust-weight ratio and propulsion times for the four missions mentioned in the INTRODUCTION should be indicated.

For the satellite-sustainer mission, thrust-weight ratios of the order of 10^{-5} or 10^{-6} are adequate for overcoming the drag at altitudes down to about 100 miles. The propulsion time, of course, depends on the purpose of the satellite and is limited only by the requirement that it cannot be so large that the payload becomes zero. For the orbit-control application, values of F/W_0 of the order of 10^{-5} or 10^{-6} are again adequate if fast maneuvers are unnecessary. The propulsion time again depends on the intended application.

For lunar and interplanetary voyages the required propulsion time is found by integrating the equation of motion of a vehicle propelled by a constant thrust in a gravitational field. Calculations were made for several values of the thrust-weight ratio F/W_0. A typical result is shown in figure 1. The trajectory is that followed by a vehicle propelled by a constant thrust, starting from a satellite orbit near the Earth. The initial thrust-weight ratio is 10^{-4}. The vehicle follows a spiral path, with very gradual increase in distance from the Earth at first. As the gravitational field becomes weaker, the speed of recession from the Earth increases. After about 80 days, the orbit of the moon is approached. A high-thrust rocket would reach the moon's orbit in about 3 days. Thus, thrust-weight ratios of the order of 10^{-4} or less are not desirable for journeys to the moon. If the thrust-weight ratio is increased to 10^{-3}, the moon is reached in the more satisfactory time of about 8 days, but the required powerplant weight for $F/W_0 = 10^{-3}$ is considerably lower than appears possible with the electric propulsion systems that now appear feasible.

If the thrust corresponding to $F/W_0 = 10^{-4}$ is applied for about 47 more days (about 127 days in all), the vehicle acquires enough energy to follow the least-energy transfer ellipse from Earth's orbit to Mar's orbit. This trajectory is shown in figure 2. Also indicated in this figure are the times required to accomplish each part of the journey with an initial F/W_0 of 10^{-4} and with a high-thrust rocket. Because of the long wait required at Mars before the Earth and Mars are in a favorable position for the return trip, the total time required for the round trip with low thrust is comparable with that required with impulse rockets. The time difference is due entirely to the larger time required by the low-thrust device to spiral out of, and back into, a satellite orbit near the Earth's surface. [The round-trip time for the low F/W_0 is actually somewhat less than that shown (perhaps 1150 days instead of 1205), because the mass of the vehcile is less when it returns to the

Earth than when it left. This mass reduction, due to consumption of propellant and subsistance supplies, was ignored in the time computations.]

It is possible with the low-thrust device to apply thrust continuously in going from Earth's orbit to Mars' orbit. The transit time is thereby decreased, but the wait time at Mars is correspondingly increased. Time saving is not great for $F/W_0 = 10^{-4}$, and the required powerplant and propellant weight is considerably increased. If a thrust-weight ratio of about 10^{-3} can be achieved, however, the required time might be considerably reduced. By applying thrust continuously with this thrust-weight ratio, the transit time to Mars is reduced sufficiently that only a few days must be spent on Mars before a favorable return time appears. In this manner, the total duration of the trip can be reduced to about 170 days. As mentioned before, however, the attainment of the low specific powerplant weight required for this thrust-weight ratio seems unlikely.

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When the propulsion time for a mission has been calculated, the payload ratio available for the mission can be obtained from equation (7) for suitable values of F/W_0 and α. The equation is plotted in figure 3. For the low thrust-weight ratios required for satellite control or satellite-sustainer missions (of the order of 10^{-5} or 10^{-6}), the allowable powerplant specific weights are quite large, even for very long sustaining times. For the propulsion time required for the Mars journey, however (indicated by the circle points on each curve), specific propellant weights of the order of 20 pounds per kilowatt or less are needed in order to make the journey with $F/W_0 = 10^{-4}$ with significant payload ratio. With lower F/W_0, the required propulsion times become much too large.

The previous discussion has assumed that values of the specific impulse near optimum can be achieved. The effect of nonoptimum specific impulse on the payload ratio for the Mars round trip is shown in figure 4 for $F/W_0 = 10^{-4}$ and for three values of α. For specific powerplant weights of the order of 10 or 20, the range of specific impulses that permit near-maximum payload ratio is rather narrow. The decrease at high specific impulse is due to the large powerplant weight required, and the decrease on the low-impulse side is, of course, due to the larger propellant weight required.

For smaller α, of the order of 1, the specific impulse is not seriously limited on the high side, but the minimum allowable value is still about 5000 seconds. The fact that the minimum value is so high may seem surprising at first, since chemical rockets can accomplish the Mars mission with much lower specific impulses if enough initial weight is provided. There are two reasons for the difference. The first is that the electric systems have little staging possibility, since the heavy powerplant must be carried throughout the trip. The electric propulsion

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system is effectively a single-stage vehicle. The second reason is that the low-thrust vehicle spends a much larger time than the impulse rocket in working out of strong gravitational fields. This means that the energy expended for a given mission is greater than for the impulse rocket, and that the "characteristic velocity" for a given mission is correspondingly higher. It is easily verified that the minimum specific impulse required to attain a given energy in a gravitational field with a single stage increases as the thrust-weight ratio decreases.

To summarize the results of this preliminary discussion: Extremely high specific impulses are not desired, because the powerplant weight becomes too large. In fact, for most applications, the optimum specific impulses will lie in range from 10,000 to 30,000 seconds. Electric propulsion systems are capable of performing the round-trip Mars mission with sizable payload if specific powerplant weight is near 20 pounds per kilowatt of jet power or less, and if specific impulses near optimum are attained. Values of F/W_0 of the order of 10^{-4} are satisfactory for the Mars mission but are too small to accomplish the moon mission in reasonable time. Electric power required will depend on the desired size of the vehicle, but will be of the order of 200 kilowatts to 20 or 30 megawatts. For satellite sustainers, satellite orbit control, and auxiliary power, the electric power required will be of the order of a few kilowatts. The specific powerplant weight allowable is much higher for these applications because of the much lower thrust-weight ratios required (of the order of 10^{-5} to 10^{-6}).

ELECTRIC POWER GENERATORS

The various possible combinations of the basic energy sources, electric power generators, and thrust generators listed previously will now be considered. For convenience, the feasible combinations of basic energy sources and electric power generators will be discussed first.

Chemical Batteries

The nonelectric propulsion schemes will need small, lightweight auxiliary electric power sources for instruments. The familiar sources for applications of this type are chemical batteries. The all-important factor for flight applications is the ratio of weight to power. Two disadvantages of today's chemical batteries are that these batteries are basically low voltage sources and that they can be as heavy as lead. The question is whether either of these unfavorable features can be overcome by research.

As to the voltage problem, the laws of thermodynamics indicate that chemical cell will always be a low voltage source (i.e., less than 5 v).

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Higher voltages may be obtained by series grouping of cells. Therefore, the reliability of the individual cells will probably limit the total voltage of a chemical battery to a value of the order of 1000 volts.

In general, the time during which a battery must supply power determines its weight. The ratios of weight to power for a few commercial batteries at various load times are compared in figure 5. The ordinate showing this ratio has the units of pounds per watt; most of the later figures use kilowatts as a basis. The familiar lead-acid and common dry cells do not show up well in the weight comparison.

The mercury (Ruben) cell was developed during World War II for "walkie-talkie" radios because of its favorable weight and compactness. The mercury cell has been considered for the project Vanguard satellite auxiliary power source because of its insensitivity to pressure and its temperature range (-65° to 250° F).

The silver-zinc-alkaline cell is another newcomer that has gained popularity for missile applications. It can supply near-rated ampere-hour capacity at great overloads, but it has poor temperature characteristics.

Research on fuel cells has been carried on for over 50 years, but the first commercial venture is the National Carbon H_2-O_2 cell now being used by the services for remote radar stations. It is very advantageous for long and continuous service. Unlike other cells, the electrodes of the hydrogen-oxygen cell are permanent; similarly, the liquid electrolyte needs only occasional care. These features, together with gaseous reactants that are easily fed continuously into the cell, result in a system that is uniquely suited for service over long times. For times over about 100 days, the major weight will be the hydrogen-oxygen gas containers. Theoretically, 1350 ampere-hours can be obtained from the reaction of 1 pound of gases; this is probably the limit obtainable from any chemical reaction. The H_2-O_2 fuel curve of figure 5 reflects a 4-pound container storage weight penalty for every 1/9 pound of H_2 and 8/9 pound of O_2. These gases were assumed to be stored in liquid phase for long times in containers only slightly lighter than those used today for ground storage. Obviously, some development work on this H_2-O_2 fuel cell would be required to develop a system suitable for space propulsion, but for long-time service, this cell will be the best of the chemical batteries.

Other Low-Power Electric Sources

As can be seen from figure 5, the weight of chemical batteries is quite high. To achieve greater savings in weight it is desirable to

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carry a more compact energy source or to tap some external energy source. The first of these possibilities involves the use of nuclear energy. The specific energy (Btu/lb) is from 10^5 to 10^6 times greater for nuclear reactions than for chemical reactions. In the low power range (100 to 1000 w), the decay energy arising from radioisotopes appears attractive. By comparison, small nuclear reactors needed to produce power in this range would be heavy.

The second of the weight-saving possibilities involves external energy sources such as solar energy. In figure 6 is plotted pounds per watt against time in days for some proposed lightweight electrical powerplants in the 100- to 1000-watt category using a radioisotope source or solar energy. The radioisotope chosen, mainly because of its high specific energy and potential availability, was Po^{210}, an alpha-emitter. The H_2-O_2 fuel-cell curve from figure 5 is also shown in figure 6 for comparison.

The two horizontal lines of figure 6 represent the solar battery, the lower for the solar battery in the sun full time, the upper for the solar battery in the sun half time. Most of the weight difference is due to the batteries required to store electricity for the times when the latter system is not in the sun.

The other systems shown in figure 6 are all radioisotope systems. The first system to be considered is the thermopile. In this system, studied by The Martin Company (ref. 1), the heat produced by the radioisotopes is used to induce an electric current in a thermopile with lead sulfide and zinc-antimony alloy elements. Most of the weight of this system is in the thermopile itself, and therefore a thermopile system using solar energy would not yield substantially better weight-power ratios. A second system studied by The Martin Company (ref. 1) uses radioisotope heat to boil mercury. The mercury vapor turns a turbine driving an electric generator. The curve for this system falls well below the one for the thermopile system.

A third system, studied at the NACA Lewis laboratory, utilizes radioisotope energy in a different fashion. In this system (fig. 7) the alpha particles dissociate water into hydrogen and hydrogen peroxide in the decomposition chamber. The hydrogen, being a gas of low solubility, separates from the water stream. The hydrogen peroxide is carried by the stream to a second chamber where it passes over a catalyst and decomposes to oxygen and water. The oxygen is removed at this point. The two gases, hydrogen and oxygen, are fed into a fuel cell similar to the one developed by the National Carbon laboratory. In this cell the gases react to give water and electrical energy. Since the radiolytic process makes only partial use of the available energy, much of the radioisotope energy goes into heat that must be rejected from the system. The water stream is therefore passed through a radiator to remove this heat.

The radioisotope-fuel-cell system falls on about the same curve as the radioisotope - mercury-vapor system (fig. 6). However, the radioisotope-fuel-cell system shows promise of efficiency improvement through the use of semiconductor materials as intermediates in the water decomposition process (ref. 2). The possibility of a twofold or even a fourfold increase in efficiency appears good. As can be seen from figure 6, the radioisotope-fuel-cell system sensitized to twice the unsensitized efficiency gives a curve that falls below all except the solar battery in full sun up to a period of $1\frac{1}{2}$ years. 4793-II

For long periods of time the solar-energy systems appear to be best on a weight basis. For shorter times, the mercury-vapor and fuel-cell systems appear to be the best, the fuel-cell system showing potential for future improvement.

Nuclear-Electric Powerplant

Fission of uranium is considered as the energy source for turboelectric powerplants of 500- to 20,000-kilowatt electric output. A specific configuration is selected in order that weight can be estimated for use in the propulsion study. This does not imply that either the specific configuration or the weights have been optimized but only that they are specific.

A turboelectric powerplant could be arranged as shown in figure 8(a). If the working fluid is a gas, the gas could be heated in the reactor, expanded in the turbine, cooled in the radiator, and compressed by the compressor to its initial pressure, thereby completing the cycle. In space, heat must be rejected from the radiator by thermal radiation rather than by convection, because there is no air to act as a heat sink. If the fluid entering the reactor is a liquid, the liquid could be boiled by the heat addition in the reactor. In this case, the resulting vapor would be condensed in the radiator, and the compressor would be replaced by a pump.

Shielding of the reactor is required in order to protect the crew. If the cycle's working fluid becomes radioactive on passing through the reactor, the shielding problem is considerably complicated, because all components of the cycle (the turbine, the radiator, and the pump) also then release radiation requiring shielding. This activation of the working fluid can be avoided by introducing an intermediate heat exchanger as shown in figure 8(b). One fluid passes through only the reactor and the intermediate heat exchanger. Another fluid is heated in the intermediate heat exchanger and used as the cycle's working fluid. In this way, the turbine, the pump, and the radiator do not become radioactive.

Since original studies showed the radiator to be very large, ways of reducing radiator size were investigated. The variation in radiator area with radiator temperature is shown in figure 9, where two classes of working fluid are compared for a single turbine-inlet temperature of 2040° F. The helium curve indicates what can be accomplished by using gases. The large compressor work penalizes gas cycles and requires a low temperature entering the radiator. As a consequence, the radiator areas per kilowatt of electrical output are large for helium. The two vapor cycles shown are comparable in radiator area, sodium being a little better because of its higher critical temperature. The attainable radiator areas within a given temperature limit are better by more than an order of magnitude for vapors than for gases. The remainder of this study therefore considers only vapor cycles. For sodium, a radiator area of 0.8 square foot per kilowatt is required for a radiator temperature of 1340° F.

For the temperature shown, sodium is superior to mercury because of the pressures involved. At 2040° F, mercury boils at 5400 pounds per square inch. At 1340° F, mercury condenses at 900 pounds per square inch. These pressures will add to the powerplant weight by requiring heavy walls. The cross section of sodium for capture of thermal neutrons is also considerably superior to that of natural mercury. This disadvantage of mercury could be largely eliminated by isotopic separation of the mercury, but sodium was chosen as the more promising working fluid for further study.

The pressures and temperatures of a sodium cycle are shown in figure 10 along with a schematic arrangement of the powerplant. Liquid sodium at 2340° F circulates through the reactor and the heat exchanger. The absence of oxygen in space will help to permit operation at these temperatures. In the heat exchanger, the cycle working fluid is heated to 2240° F. The pressure of 200 pounds per square inch is sufficient to keep the sodium a liquid even at 2340° F. This liquid sodium then enters the evaporator. At 2040° F and 70 pounds per square inch, $3\frac{1}{2}$ percent of the sodium leaves as a vapor. The sodium vapor expands in the turbine to 2.7 pounds per square inch, producing about 400 Btu from each pound of sodium passing through the turbine. The over-all cycle efficiency is 20 percent. Condensation of vapor in the radiator presents a problem in removing liquid from the tube walls that does not exist on Earth because of the gravitational field. The whole powerplant could be rotated about a longitudinal axis in order that centrifugal force could keep the condensed liquid moving along the walls of the radiator. This rotation would also provide an artificial gravity field for the crew.

The estimated weights of such a powerplant are shown in figure 11. At 20,000 kilowatts, the radiator has the dominant weight, in spite of the fact that the design was varied to minimize this weight. The generator and reactor make significant but small contributions. The miscellaneous item includes the heat exchanger, pumps, turbine, evaporator, piping, sodium, and structure, none of which individually adds much to the weight. At 20,000 kilowatts, the shield weighs about 1 pound per kilowatt. As the design value of power changes, the weight per kilowatt of most of these items remains essentially constant. The shield is an obvious exception. As the design power changes, shield weight changes slowly, with the result that its weight per kilowatt climbs steeply as power goes down.

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At 20,000 kilowatts, the estimated total powerplant weight is $5\frac{1}{2}$ pounds per kilowatt. For the low-power end of the power spectrum, reference 3 concludes that a powerplant weight of 160 pounds per kilowatt is attainable with an electric output of 3 kilowatts.

Weight estimates such as that in figure 11 must be predicated on some presumed geometrical configuration. The geometry considered in the weight estimation for figure 11 is shown in figure 12. For 20,000 kilowatts, the over-all length is 600 feet. The radiator dominates in terms of physical size as well as weight. The crew compartment is separated from the reactor in order to reduce the shielding requirements. In order to tie the whole device together, a $2\frac{1}{2}$-foot-diameter tube is provided that will take both tension and compression; such a structure will keep the crew compartment away from the reactor. This tube also has some strength as a beam and will supply some stiffness to the whole vehicle.

The reactor and turbine ends of the powerplant are shown in somewhat more detail in figures 13(a) and (b), respectively. The problem of shielding the crew from the reactor is simplified in outer space because of the absence of any air for scattering. For this reason shadow shielding was used for the crew compartment, the radiator, and all of the machinery in the shadow of the shield. The shielding was designed without regard for the less well-known effects of cosmic radiation. A literature survey indicates that rather heavy crew shielding is required to protect against cosmic radiation. In spite of such shielding of the crew compartment, it appears that shielding of the reactor will still be required.

Evaporation of liquid into vapor in the absence of a gravity field presents a problem of separating the two phases similar to that encountered in the radiator. Location of the heat exchanger and the evaporator near the axis of rotation of the vehicle keeps low the centrifugal acceleration within these pieces of equipment. For this reason, the 130-psi drop across the evaporator was exploited to produce a rapid rotary movement of the liquid sodium within the evaporator. About 1000 g's of radial acceleration are available for separation of the vapor from the liquid.

The number of turbine stages is sensitive to the allowable centrifugal stress in the turbine blades. Two stages are shown for the turbine in spite of the work requirement of 400 Btu per pound.

Radiator weight was kept low by assuming that the radiator could be built of tubes having a wall thickness of 0.020 inch. Walls of this thickness are susceptible to damage by meteoroids. Reference 4 indicates that, on the average, such a radiator for a 20,000-kilowatt powerplant will suffer one penetration by a meteoroid each 40 days; only one in 10,000 such holes will be bigger than 1/4 inch. The radiator is segmented in order that valves in the manifolding can isolate the damage from a meteoroid until the resulting leak can be repaired. Thicker walls for the radiator should decrease the incidence of damage by meteoroids; a 0.025-inch wall thickness will increase the average time between penetrations to 100 days, and it will increase the powerplant weight by 5000 pounds, or 5 percent. Damage by meteoroids cannot be avoided with certainty because of the extreme penetration of rare particles. Other estimates of damage by meteoroids disagree by an order of magnitude (ref. 5, e.g.).

In summary, this hypothetical powerplant has three salient features: (1) The working fluid is a vapor; (2) the radiator is very light in construction, depending on an ability to recover from meteoroid damage; (3) the operating temperatures are fairly high. Failure to incorporate these three characteristics will result in a big increase in powerplant weight.

Solar Turboelectric System

About 100 watts of solar radiation is incident on a square foot of surface normal to the sun at the Earth's orbit. A possible scheme for using this energy is shown in the block diagram of figure 14(a). The circuit is identical with that just described for the nuclear-electric system, except that the reactor is replaced by a very large mirror that focuses the solar energy on a heat exchanger. (The present system is proposed rather than one using thermopiles, because the thermopiles make the weight much greater.)

The main problem in any solar system is the mirror. A possible arrangement is shown in figure 14(b). The mirror is a large polyester balloon, as proposed in reference 6. Half is transparent and half is silvered. The heat is focused on a heat exchanger that extends along the axis from the mirror to a point half way from the axis to the mirror. The remainder of the cycle is assumed to be exactly the same as in the nuclear-electric system. The rotating machinery is placed inside the balloon to limit the length of hot lines and also for stability. The crew compartment would also be there. Thrust chambers would be outside, as would controls for aiming them. In addition, the sphere must be separately controlled so that the mirror always faces the sun.

[redacted]

To get 20 megawatts of electric power, a balloon diameter of about 1260 feet is needed. Such a balloon, made of 1-mil-thick Mylar, weighs about 36,000 pounds. To obtain a total weight estimate, the same weights are used as for the nuclear-electric system, making allowance for controls. Then the total weight for a 20-megawatt electric power output is about 110,000 pounds, which is virtually the same as for the corresponding nuclear system. If, for some reason, a rigid mirror, not a balloon, is desired, the weight will be very much greater. If a number of small balloons replaced the single large one, a lesser weight penalty would be involved.

The main advantage of this scheme over the nuclear-electric system is that no shield is needed. Hence the equipment is readily accessible. This also means that at lower powers a weight advantage should occur because most of the components will scale more or less linearly. For example, it is estimated that a 200-kilowatt electric power system would weigh on the order of 1500 pounds.

On the other hand, there are serious difficulties. The power available varies as the square of the distance from the sun. The power at Mars is about 40 percent of that at Earth. Another problem is that near a planet the vehicle may be shielded from the sun's rays. Then no power is delivered and large storage facilities may be needed. Finally, the size makes meteor damage more likely. Although repairs are simple and very little gas is required to inflate the balloon, this problem might well make its use impossible. In such a case, a rigid and unfortunately a heavy mirror would be required.

The use of this system for a satellite sustainer is improbable, as the balloon probably cannot overcome its own drag at altitudes of less than about 300 miles. It might, however, be used at higher altitudes for orbit control or auxiliary power. For such applications, about 3 kilowatts of electric power might be obtained for about 300 pounds weight, very little of which is in the 15-foot mirror.

Nuclear Fusion

Both fission and solar power have been considered. Perhaps fusion will someday have a place in this type of application.

There is little to gain by using fusion energy as a heat source for a thermodynamic cycle with a working fluid. Such a system would be very similar to the fission reactor system that uses a sodium-vapor cycle and involves heat exchangers, radiators, and turbines. The thermonuclear machine would merely replace the reactor of the fission system, and the reactor is only a small portion of the weight of that system. Therefore, there is no particular advantage to a thermonuclear machine used strictly as a means of heating a working fluid.

[redacted]

Both the direct production of electricity and the direct production of thrust should be possible with a thermonuclear machine and probably could be realized in the future. The weight of such a system is difficult to estimate. Although thermonuclear theory is not far advanced, a few observations can be made.

All current thermonuclear machines of interest utilize one of two types of magnetic fields for containing the high-temperature plasma, a magnetic field produced by external field coils or a magnetic field induced by high currents in the plasma itself. The principal weight associated with the first is the weight of the field windings. At present, large-volume magnetic fields can be wound with field strengths of the order of 50 kilogauss; and with fields of this strength the weight of the thermonuclear machines would be much greater than those of the fission or solar systems discussed previously. Production of higher field strengths is being investigated; and, if field strengths of the order of 200 kilogauss or higher can be obtained, then some of the machines using externally wound fields might become of interest.

The principal weight associated with the machines that rely on a current-induced magnetic field is the weight of the condenser bank used to produce the high plasma current required. Recent advances in the use of mixtures containing barium titanate as a dielectric material offer the hope that this weight can be reduced to manageable proportions. A rough estimate was made of the weight per jet kilowatt of a machine using a current-induced magnetic field for confinement. A stabilized pinch machine was considered, and weights were estimated for its various components such as the main condenser bank, the stabilizing field condenser bank and coil, the preheating or "collapse" field condenser bank, the vacuum system, the neutron shield, and the cooling system. These estimates are very uncertain; but, if such a system can be made to work, the weight per jet kilowatt might be of the order of 3 pounds, or the thrust-weight ratio of the order of 8×10^{-4} for powerplants of the larger sizes being considered.

Comparison of Electric Power Generators

Most of the feasible combinations of basic energy sources and electric power generators have been discussed. A comparison of the more promising ones is shown in figure 15, in which the estimated weight of several systems is plotted against the electric power output. Shown by the bars near the top of this figure are the ranges of power required for the missions being considered. The nuclear turboelectric system without shielding and the solar turboelectric system in the sun full time are comparable in weight throughout the power spectrum. For auxiliary power and the satellite sustainer and control applications, a number of systems are competitive. The weight of the solar battery and solar turboelectric systems will depend greatly on the penalty in weight necessary for part-time-in-sun operation. For propulsion applications, this penalty need never be more than twice the weight for full time in sun, since the mission of interest can be performed by applying proportionately greater thrust during the time that the vehicle is in the sun.

For the higher power levels required for lunar and Mars missions, the only systems that remain competitive are the nuclear turboelectric, solar turboelectric, and the fusion-electric generator. With increasing power, the shielding weight becomes a less significant percentage, and both the nuclear turboelectric and the solar turboelectric systems approach a linear variation of weight with power. The straight line at higher powers corresponds to a slope of about 5 pounds per kilowatt. If the electric power could be converted into jet power with high efficiency and with little additional weight in thrust-generator apparatus, this value would represent approximately the specific powerplant weight α and would be a very satisfactory value for the Mars mission. It is shown later that the required additional weight of the thrust generators will be moderate for ion or plasma accelerators, so that the value of α will depend principally on the efficiency of conversion of electric power to jet power.

The single weight estimate for the fusion-electric system indicates a weight of about 3 pounds per kilowatt at 20 megawatts. The curve of weight against power output will probably be more nearly horizontal for this system than for the fission turboelectric system; consequently, it appears that the fusion-electric system might be applicable chiefly to large-scale expeditions to Mars and beyond.

THRUST GENERATORS

Electric-Arc Chambers

One method of generating thrust with electric power is to heat a propellant with an electric-arc discharge. This method is illustrated in figure 16. The arc chamber is similar to a rocket rotor, the principal difference being that the propellant is heated electrically instead of chemically. An arc is struck from the anode to the nozzle walls, which serve as the negative electrode. The propellant can be passed along the combustion chamber and the anode to provide regenerative cooling. The propellant then passes through the electric arc, where it is heated, and expands through the nozzle. If the power level is so high that the propellant does not provide sufficient cooling, another cooling circuit must be provided, and the excess heat must be rejected through radiation.

There are three basic limitations to the specific impulse attainable with an electric-arc chamber: (1) electrode consumption rate, (2) local heat-transfer rate at the throat, and (3) over-all cooling rate. An estimate based on electrode consumption rate for an uncooled graphite anode indicates that the maximum specific impulse for this case may be limited to about 1500 seconds. This limit arises when all electric power goes into vaporizing the electrode and none goes into the propellant. The equations that determine this limitation, and the estimated maximum specific impulses and consumption rates for several electrode materials are shown in table I. The consumption rates for the materials other than graphite were estimated from an analysis based on vaporization enthalpies.

If a method is found for overcoming the limitation due to electrode consumption, the cooling requirements impose further limitations. The throat heat-transfer rate becomes severe at high chamber pressures but can be brought down to reasonable values by decreasing the pressure. However, going to low pressures increases the fraction of the total enthalpy that must be removed to maintain a given maximum allowable surface temperature throughout the arc chamber and nozzle.

The effect of over-all cooling requirement on specific impulse is shown in figure 17 for arc chambers designed to produce 1 pound and 100,000 pounds of thrust. These curves are for 10-atmosphere chamber pressure and for a nozzle that produces 75 percent of the maximum specific impulse. If a larger nozzle or a lower pressure is used, the limitation on maximum specific impulse becomes more severe. If a higher pressure is used, the throat heat-transfer rate becomes excessive. Figure 17 shows that, as electric power is increased, specific impulse increases almost linearly first but then reaches a maximum value. At this maximum, further increases in power must be removed by cooling to maintain the allowable surface temperature. The maximum specific impulse for the low-thrust nozzle is about 2000 seconds and for the high-thrust nozzle about 4000 seconds.

If the powers required to attain these specific impulses are considered and the corresponding powerplant weights from figure 15 are used, the thrust-weight ratios for the electric-arc chamber for high and low thrust level are of the order 10^{-3} to 6×10^{-4} in the high-specific-impulse range. At lower specific impulses and with larger nozzles, the thrust-weight ratio might range up to about 10^{-2}. Thus, the electric-arc-chamber propulsion system is incapable of takeoff or satellite launching application. Furthermore, for such low thrust-weight ratios, the specific impulse of about 4000 seconds is not particularly high. Figure 4 showed that, for a thrust-weight ratio of 10^{-4}, at least 5000 seconds are needed to undertake the Mars round-trip mission. For F/W_0 of 10^{-3}, the corresponding minimum is of the order of 4000 seconds. The electric-arc-chamber propulsion system is therefore marginal for this mission.

A possibility not yet mentioned is that, at higher temperatures, for which the ionization becomes more significant, magnetic fields might be used to keep the hot propellant away from the surfaces. The cooling limitation on specific impulse might thereby be alleviated. This possibility has not been examined in detail, but additional weight in field coils and additional electric power would certainly be required. Furthermore, other electric thrust generators are capable of achieving higher specific impulse and comparable thrust-weight ratios without going through the propellant heating cycle. It may therefore be concluded that the electric-arc chamber is not a very promising method of generating thrust from electric power.

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Thrust from Ion Acceleration

The arc system suffers from overheating. A scheme that does not have a real heat difficulty is one in which ions are generated and then accelerated electrostatically. Figure 18 shows the parts of such an outfit. The propellant is ionized and then accelerated electrostatically and finally exhausted to space. The accelerator will require about 20,000 volts of direct current. The power-generating systems discussed earlier were designed on the basis of low-voltage alternating current. However, if a high-voltage a-c generator is used and rectifiers are added to get direct current, only a minor weight penalty is incurred.

The items shown in figure 18 will not give any thrust at all. If only positive ions are emitted, a space charge will immediately build up outside the ship. This effective decelerating potential will immediately stop all flow. Hence, electrons should be emitted at the same rate to neutralize the charge. Fortunately, this is not difficult. The main design problem associated with this space charge is that charge neutralization must occur in a very short distance - of the order of a fraction of an inch - if reasonable current densities are desired.

A second design problem is that the ionization chamber must be simple and able to operate for long periods. In addition, it should ionize all of the propellant. What is not ionized will not be accelerated to very large velocity and so will be wasted.

One especially simple ionization scheme has been suggested. The alkali metals (cesium, rubidium, and potassium) have low first-ionization potentials - about 4 electron volts. On the other hand, the work function of heated platinum or tungsten is larger than this. Experiments conducted about 30 years ago (refs. 7 and 8) showed that, when such an alkali vapor is passed over a suitably heated tungsten plate, the atom is adsorbed and reemitted as a positive ion. If the plates are at about 1800° F, the probability of ionization is virtually 100 percent (ref. 9).

Figure 19 shows a system using this ionization method. The arrangement is due to Stuhlinger (ref. 10), as is the idea of using this method of generating ions. A vapor of cesium is admitted and passed over a series of heated plates that are maintained at a small potential difference. Beyond the plates, a large potential difference is maintained to accelerate the ions to the desired final velocity. They are emitted to space as are electrons to neutralize the space charge.

It is desirable to have as large a current density in the jet as possible, because the total current may be as much as 500 amperes or more. One severe limitation is the external space charge:

$$\text{Limiting current density} \propto \frac{1}{(\text{Distance})^2}\sqrt{\frac{\text{Charge}}{\text{Mass}}(\text{Voltage})^3} \qquad (8)$$

The allowable current density varies with the charge, mass, and voltage through which the ions have been accelerated; but most important, it varies inversely as the square of the distance to neutralization. For example, for cesium at an impulse of 15,000 seconds, the required accelerating potential is 10,000 volts. If neutralization takes place in 1 inch, a current density of only 1 ampere per square foot is allowed. This can lead to exhaust areas of the order of 500 square feet.

Now if the total current required is known in terms of the thrust and impulse, then the required jet area can be written as follows:

$$\text{Jet area} \propto \frac{\text{Thrust}}{(\text{Specific impulse})^4}\left(\frac{\text{Charge} \times \text{Distance}}{\text{Mass}}\right)^2 \qquad (9)$$

What is wanted is heavy ions, singly charged. This is fortunate, because the single charge is the most easily obtained. Also desired are rapid neutralization and high specific impulse.

There are two main reasons for wanting low jet area. One is concerned with thermal radiation loss, which could be exorbitant. The other is that the weight will increase with the area. If it is arbitrarily assumed that the weight is proportional to this area, some weight estimates can be made for a nuclear-electric system flying to Mars. Table II shows the results for a Mars mission. The first two cases correspond to 4 pounds per kilowatt at 15,000 seconds impulse, with variation as in the area equation (9). Efficiencies of 40 and 80 percent are assumed. An optimum impulse of about 20,000 seconds is found, and the propulsion-system weights are on the order of 6000 to 8000 pounds per pound of thrust. If the ion source weighs nothing at all, slightly smaller results follow. If the current density is not governing, then the area equation will not set the weight. Then it might vary linearly with impulse. Such a case is shown in the last line of table II. Two general results follow from these estimations. Regardless of the assumptions concerning the weight of the ion accelerator, the optimum impulse is in the range 14,000 to 23,000 seconds for all cases. The required accelerating potential will therefore be under about 35,000 volts. The weights all show that a thrust to gross vehicle weight ratio of 10^{-3} is not possible but that 10^{-4} is attainable. This means that the Mars mission can be accomplished in reasonable time.

If the jet can be neutralized very rapidly (say, in 1/10 in.), then high current densities are possible. Then the possible rate at which ions can be generated limits the size. In this case the contact method of ion generation may not be satisfactory. However, if electron bombardment is used, very high current densities can be obtained (on the order of several hundred amp/sq ft). However, this method has a certain complexity and may have lower ionization efficiency. Considerable study of such systems is in order.

Plasma Accelerators

As a result of large space charges built up at the accelerator exit, the cross-sectional area of the ion accelerator is large. If ions and electrons were both accelerated in the same direction and charge neutrality preserved, then much higher particle densities and much smaller accelerators would be possible. An ionized gas in which charge neutrality is preserved is called a plasma, and the associated accelerator might be called a plasma accelerator. Since electric fields tend to accelerate oppositely charged particles in opposite directions, magnetic fields must be used to accelerate plasmas.

One idea for a plasma accelerator that has received a considerable amount of experimental work is shown in figure 20. This particular plasma accelerator was devised by W. H. Bostick (ref. 11). Sketch 1 of figure 20 shows two electrodes in an insulator material. A condenser is discharged through these two electrodes to produce an arc. In space the ions and electrons in the arc would come from the electrodes. The three sketches of figure 20 represent three different stages in the arc discharge. The time interval between sketches is of the order of a fraction of a microsecond.

The current in the arc induces a magnetic field as shown. The magnetic field and current interact to produce a force in a direction perpendicular both to the current and to the magnetic field. The magnetic field is stronger on the inside of the curved arc, and therefore there is a net force in the outward direction which accelerates the plasma as shown. Specific impulses up to about 20,000 seconds have been measured.

Figure 21 shows a Kerr cell photograph (taken by Bostick) of a plasma about 1/2 microsecond after firing. The bright spot is at the accelerator, and the luminous area has the typical horseshoe shape shown in sketch 2 of figure 20.

A propulsion system using this plasma accelerator is shown in figure 22. Several of the plasma accelerators, each with its own condenser and switch, are connected in parallel to a high voltage d-c source. The system is designed so that each accelerator fires at the rate of 1000 pulses per second. The thrust from such a system would be about 1 pound per 100 plasma accelerators. The weight per kilowatt of jet power is only a small fraction of a pound for the propulsion portion of the system. The principal weight in the system would be the weight of the electric generating system. If a nuclear-electric system were used, the weight would be about $5\frac{1}{2}$ pounds per kilowatt electric power. As discussed previously, the weight penalty of the high-voltage d-c supply is small.

The plasma accelerators are not nearly 100 percent efficient, as there are unavoidable losses in heating the electrodes and in the switch. Some preliminary experiments carried out at the Lewis laboratory indicate that efficiencies of the order of 40 percent or higher could be attained. With an efficiency of 40 percent, the optimum specific impulse for the Mars journey would be about 11,000 seconds, and the weight of propellant and powerplant per pound of thrust about 7000 pounds. If an efficiency of 80 percent could be attained, these figures would become about 15,000 seconds and 5000 pounds, respectively. These values are about the same as those for the ion-accelerator propulsion system.

Photon Generators

The use of artifically generated photons is often referred to as the ultimate in jet propulsion. As yet, however, no satisfactory method of generating photons is known. If the electrical systems discussed are relied on, the specific impulse of photons is much too high. Even with 100-percent efficiency of conversion of electric power into directed photons, a nuclear-electric system of the sort considered would weigh about $3\frac{1}{2}$ million pounds to generate 1 pound of thrust. This gives a thrust-weight ratio of the order of 10^{-7}, and about 10^{-4} is needed to get to Mars in a reasonable length of time. For journeys within the solar system, therefore, it is much better on an initial weight basis to limit specific impulses to the order of 30,000 rather than 30,000,000.

For interstellar or intergalactic journeys, photon propulsion may be the ultimate solution, but a process must first be found to convert large portions of mass into directed photons. Current fission and fusion reactions contemplated for power generation convert only about 5 percent of the mass into energy.

Photon and Radioisotope Sails

Since it seems uneconomical to try to produce photons, perhaps the photons provided by the sun could be used. A perfect reflector normal to the suns rays, outside the atmosphere, at the same distance from the sun as the earth, feels a force of about 2×10^{-7} pound per foot. This is a small force, and accordingly requires lightweight reflectors. Figure 23 shows a plastic balloon that is silvered over the outside surface and has instruments located in the center. If the balloon were 1/2-mil-thick plastic, its weight per square foot of surface would be 3×10^{-3} pound per square foot. The ideal thrust-weight ratio, that is the thrust-weight ratio for a section of surface normal to the sun's rays, would be about 7×10^{-5}. The actual thrust-weight ratio would be less than half of this, since half of the balloon surface is always inoperative and some of the operating surface will not be normal to the sun.

Actually, this photon sail would not be able to escape from a satellite orbit to free space unless there were some kind of control system that would obscure the mirror when the sun was not in the proper position. This would further add to the weight.

In view of the low thrust-weight ratio, this idea does not look too interesting for any of the manned missions being considered. The low thrust-weight ratio and the control problem make the idea uninteresting for the unmanned missions, since development of an automatic control system would require an effort that would probably be better expended elsewhere.

Another idea similar to the photon sail is the radioactive sail or alpha sail shown in figure 24. A radioisotope which is an alpha-emitter (Po^{210} in this case) is embedded in a 0.2-mil layer of plastic which is backed by a 1-mil layer of plastic. Alpha particles are emitted in both directions, but those in one direction are stopped by the 1-mil-thick plastic backing sheet. This results in a net thrust of 10^{-6} pound per square foot, which is higher than that of the photon sail, but the weight is also higher, 9×10^{-3} pound per square foot. The net result is that the ideal thrust-weight ratio would be about 10^{-4}, about the same as that of the photon sail.

A "parachute" geometry has been shown for the radioisotope sail instead of a balloon geometry as for the photon sail. Either geometry is possible for either type of sail. The problems of control are probably greater for a parachute geometry.

The alpha sail does not give any appreciably advantage in thrust-weight ratio compared with the photon sail and has at least one great disadvantage - namely, the loss in thrust with the decay of the radioisotope. Therefore, of the two schemes the photon sail looks more interesting; but, as previously observed, it is not very attractive for the missions being considered.

Recombination and Solar Ramjets

Two Earth-bound propulsion systems have been proposed over the years which, in theory, could support flight indefinitely in the Earth's rarefied upper atmosphere, without carrying chemical fuel. Discussion of the ionosphere recombination ramjet will be presented first, and then the solar-powered ramjet will be discussed briefly.

Readers interested in more details on the recombination ramjet should also see reference 12.

Recombination ionosphere ramjet. - Above 52 miles in the Earth's atmosphere the oxygen and nitrogen of air are dissociated by the sun's ultraviolet rays into chemically active free radicals or atoms. The idea of a recombination ramjet is to take these energetic air particles on board and to convert their energy into heat and thereby obtain thrust.

Granting for the moment that the idea is sound, the first question is, At what altitudes and flight speeds is the ionosphere chemical energy useful for propulsion? A preliminary analysis showed that even an all-supersonic ramjet would require more energy for providing lift and overcoming drag than is available at any ionosphere altitude. Therefore, a recombination ramjet is considered traveling at orbital velocity where only drag need be overcome.

The thrust that could theoretically be generated from the recombination energy available is compared with the drag for several nacelle configurations in figure 25. The thrust parameter is the thrust divided by the ramjet inlet area and the ambient air density. Similarly, the external drag is divided by the inlet area and density for direct comparison. The crosshatched area indicates the probable limits on the energy available at altitudes from 300,000 to 700,000 feet. The range of energies shown is the result of uncertainty in ionosphere physical properties. The external drags are shown for a ramjet length of 100 feet. The far right curve is the drag for a truncated cone with a positive angle of 2° and an inlet radius of 10 feet. Similarly, the next curve is for a more promising configuration: -4.3° angle and 10-foot radius. Finally, the far left curve is for a nacelle with a -8.6° cone half-angle and 20-foot inlet radius.

Only the -8.6° nacelle gives a drag appreciably lower than the probable maximum thrust. Therefore, an engine for this nacelle will be considered in more detail with a thermodynamic cycle.

Figure 26 summarizes the results of a cycle analysis. The nacelle geometry is shown to scale. A low ionosphere altitude of 328,000 feet was chosen for this example. At this altitude only oxygen is dissociated into free radicals. The cycle involves swallowing this energetic air and exhausting a hot recombined air jet. The cycle is illustrated in figure 27, which shows stations 1, 2, and 3 on a static temperature-pressure plot. Station 1 is the shock-free inlet station; station 2 is the internal throat station; station 3 is the nozzle-exit position. A frozen-composition compression from about 5×10^{-7} to 6×10^{-3} atmosphere brings the inlet air to a temperature-pressure condition at station 2 where it theoretically can be converted adiabatically and isothermally to chemical equilibrium. The exhaust expansion from station 2 to 3 is assumed to follow chemical equilibrium. Notice that from station 2 down to about 10^{-4} atmosphere, recombination is proceeding, causing the unusual temperature-pressure relation in this region. Station 3 is at a considerably higher pressure than ambient, because the nacelle geometry for this example does not allow full expansion. (If expansion to ambient pressure station 4 were possible, then the engine efficiency would be 85 percent.) The resulting over-all engine efficiency, shown in figure 26, is 22 percent; the thrust is an order of magnitude greater than external drag for this example.

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Actual hardware design of this engine involves at least two serious problems. The inlet requires a very large contraction ratio in a short length; but perhaps a multiple diffuser could do this job. Also, the chemical kinetics of recombination are not understood well enough today for proper internal flow design. However, if interest in a large, low-flying satellite is great enough, none of these problems appear unsolvable.

Solar-powered ramjet. - Another device attractive in principle is the solar-powered ramjet. It was mentioned earlier that the sun supplies the Earth with about 100 watts per square foot of normal surface as radiant energy. Perhaps this solar energy can be used directly for high-altitude satellite propulsion in a ramjet.

Naturally, the problem is how to get this radiant energy into the airstream for the heat cycle. A ramjet using air as a working fluid cannot absorb any appreciable fraction of this solar energy directly. Therefore, a convective heat exchanger must be used in conjunction with a solar-energy collector or lens. That is, the collector would heat a metal heat-transfer surface, and the air passing over this surface would be heated to supply thrust.

The basic problem of such a device is heat transfer. In a convective heat exchanger, the ΔT for heat transfer is the temperature of the wall minus the adiabatic wall temperature. The adiabatic wall temperature exceeds the material limit on today's metals at flight Mach numbers greater than about 7. Therfore, the solar-powered ramjet is not useful for satellite-sustaining.

Furthermore, for a Mach number of 1.5, which is the "follow-the-sun" velocity, Rosebrock and Johnston (ref. 13) concluded that, even taking an optimistic view, a ramjet engine using solar energy as an exclusive heat source at 200,000 to 300,000 feet is not feasible.

COMPARISON OF ELECTRIC PROPULSION SYSTEMS WITH CHEMICAL AND NUCLEAR ROCKETS

The more promising propulsion systems discussed will now be compared with chemical and nuclear rockets for several typical missions. Figure 28 shows the powerplant plus propellant weight as a function of required sustaining time for a satellite-sustainer application. The requirement is the production of a sustaining thrust of 0.05 pound. This thrust is adequate to overcome the drag of a 6-foot-diameter hemisphere-cylinder satellite about 30 feet long at an altitude of about 100 miles. The thrust could also be used for orbit control. The two curves for rockets, with specific impulses of 300 and 1000 seconds, include only the propellant required and not the powerplant. It is doubtful whether rockets could be designed to achieve these specific impulses at the very low thrust level indicated. Such rockets would probably be operated at higher thrust levels for short periods of time, but the over-all propellant consumption would be comparable to that shown. The electric systems require an electric output of about 10 kilowatts at a specific impulse of about 10,000 seconds to generate the required 0.05 pound of thrust. The weights shown are minimum values, without penalty in shielding or for part-time-in-sun operation. Depending on the magnitude of these penalties, the crossover points relative to the hypothetical rockets will shift toward larger sustaining times. No precise value can therefore be given for the sustaining time at which the electric systems become superior as to weight, but it is clear from the slopes of the curves that the electric systems will eventually become superior. In particular, if a satellite is to be maintained aloft indefinitely, the resupply weights are much less for the electric than for the chemical system owing to the much higher specific impulse.

Figure 29 compares the initial weight required for an unmanned one-way trip to Mars from a satellite orbit near Earth to a satellite orbit around Mars. An instrument payload of 2000 pounds is allowed. Two chemical rockets and two nuclear-electric ion systems are considered. The nuclear heat-transfer rocket has been omitted from comparison for this mission, because no estimates have been made of such low-weight nuclear motors. The comparison shows that the advanced chemical rocket

(I = 420 sec) is capable of undertaking this mission with little weight penalty relative to the nuclear-electric system.

Figure 30 compares initial weight required for a full-scale manned trip to the moon with landing and exploration equipment. The basic payload, which includes all items carried throughout the trip, is taken to be 10,000 pounds. An additional subsistence allowance of 10 pounds per man-day was considered, and a landing and exploration equipment weight of 16,000 pounds was assumed. The initial weight comparison is for the two chemical rockets of figure 29, the nuclear-electric-ion systems, and the low-pressure high-specific-impulse nuclear rocket described in paper 2. The unshielded nuclear-electric system is approximately the same weight as the solar turboelectric, so that this column serves a dual purpose. A specific powerplant weight of 10 pounds per kilowatt was assumed for the nuclear-electric system. This weight would be attainable if the conversion of electric power to jet power is accomplished with 70-percent efficiency and if the thrust generator weighs 2 pounds per kilowatt. These values, as previously indicated, appear to be attainable. The comparison shows that the largest drop in required initial weight occurs in going from the I = 300 chemical rocket to the I = 420 chemical rocket. There is little further gain in going to the nuclear rocket, because of the high motor weight and consequently the reduced staging advantage. Some additional weight reduction is possible by going to the nuclear-electric-ion system, but this system is unattractive for the moon mission because of the long travel time required. The advanced chemical rocket is therefore capable of undertaking this mission without excessive weight penalty.

The same systems are compared in figure 31 for a similar manned expedition to Mars. For this mission, the basic payload is 50,000 pounds and the additional landing and exploration equipment are 60,000 pounds. A substantial weight reduction, even over that obtained with the advanced chemical rocket, is possible for this mission by going to the nuclear rocket or the nuclear-electric-ion system. A point worth noting in this comparison is that the total initial payload, consisting of the basic payload, the landing and exploration equipment, and the subsistence supplies, is about 200,000 pounds. The initial gross weight for the nuclear rocket and nuclear-electric systems is therefore only $2\frac{1}{2}$ to 3 times the initial payload weight. This means that there is not too much margin left for reducing the gross weight of an expedition of this magnitude, and the nuclear propulsion systems considered are, in fact, very good systems for this mission.

CONCLUSIONS

Auxiliary Electric Power

Systems using solar energy (solar batteries and solar turboelectric systems) involve the least weight for power requirements up to a few kilowatts, provided almost-continuous operation in the sun is possible. If only half time is spent in the sun, a number of systems are competitive, including radioisotope hydrogen-oxygen cells and radioisotope turboelectric systems (for durations comparable to the half-life of the isotope). The solar turboelectric systems can be used only at altitudes above about 300 miles, since the drag of the required balloon collector is excessive below this altitude. The nuclear turboelectric system without shielding is competitive in this range of power, but shielding requirements, particularly for manned vehicles, may rule it out. Chemical batteries are competitive weightwise only for durations of operations of the order of a few days. The required voltage must also be considered in selecting auxiliary power systems.

Satellite Sustainers and Orbit Control

For periods of operation of the order of 100 to 200 days or less, a chemical rocket can provide the required propulsive energy without excessive weight penalty relative to electric systems. Particularly, if rapid orbit changes are required, the chemical rocket seems to be the only feasible propulsion system. For very long durations, or for permanent satellites, electric propulsion systems using solar energy or nuclear energy require less initial weight or resupply weight than chemical rockets. The solar turboelectric system is restricted to altitudes above about 300 miles, and the solar batteries are limited in the voltage attainable with a practical arrangement. Consequently, the nuclear turboelectric system with ion or plasma accelerators seems most satisfactory for this application if shielding weight can be kept low.

The recombination ramjet may be feasible for sustaining satellites indefinitely at altitudes near 60 miles if the powerplant is made sufficiently large. However, many serious questions remain concerning the possibility of designing the required short inlet with very large contraction, and concerning the magnitudes of the recombination rates.

Lunar and Mars Journeys

Many missions involving trips to the moon and Mars can be accomplished without excessive weight penalty with high-performance chemical rockets ($I \approx 420$ sec). These missions include one-way instrumented journeys to the moon and Mars, and manned trips to the moon. Electric

[redacted]

propulsion systems seem undesirable for the moon trip because of the long times required for the journey compared with those required for high-thrust rockets. For manned trips to Mars, however, electric propulsion systems require only moderately more time for the complete journey than the impulse rocket, and their advantage in initial weight becomes greater and greater as the size of the expedition increases. Of the electric systems considered, the nuclear turboelectric, the solar turboelectric, and possibly the fusion-powered systems are capable of supplying the required electric power with sufficiently low weight. Of the thrust generators considered, the ion-electric accelerator appears to be most promising on the basis of current technology.

The low-pressure, high-specific-impulse nuclear rocket is competitive with electric systems for large-scale Mars expeditions, and has the advantage of higher thrust-weight ratio. It has the disadvantages that much higher temperatures are required than in the electric systems and that hydrogen must be used to attain the required high specific impulse. The latter requirement imposes severe storage difficulties for long-duration journeys.

Another advantage of the electric system over both the nuclear and chemical rockets is the resupply advantage. Since the electric system has the higher specific impulse, its propellant replacement weight is much less than that of the chemical and nuclear rockets.

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SPECIFIC IMPULSE LIMITED BY ELECTRODE CONSUMPTION

MATERIAL	CONSUMPTION RATE, R, LB/SEC PER KW	MAX. SPECIFIC IMPULSE, SEC
CARBON	0.2×10^{-4}	1500
MOLYBDENUM	0.8×10^{-4}	750
TUNGSTEN	1.1×10^{-4}	640
COPPER	1.4×10^{-4}	570

$$H_O = \text{STAGNATION ENTHALPY} = \frac{\text{POWER}}{\text{MASS FLOW}} = \frac{P}{W_r + W_p}$$

W_p = ELECTRODE CONSUMPTION; W_r = PROPELLANT CONSUMPTION

$$I_{MAX} = 6.75 \sqrt{H_{O_{MAX}}} = \frac{6.75}{\sqrt{R}}$$

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Table I

ION SOURCE	EFFICIENCY, %	OPTIMUM IMPULSE (SEC)	PROPELLANT + POWERPLANT WEIGHT PER UNIT THRUST
4 LB/KW AT 15,000 SEC IMPULSE	80	23,000	5,700
	40	18,000	8,500
WEIGHTLESS	80	18,000	5,000
LINEAR WITH IMPULSE, 3 LB/KW	80	14,000	6,400

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Table II

CONSTANT-THRUST TRAJECTORY FROM SATELLITE ORBIT

THRUST/WEIGHT = 10^{-4}

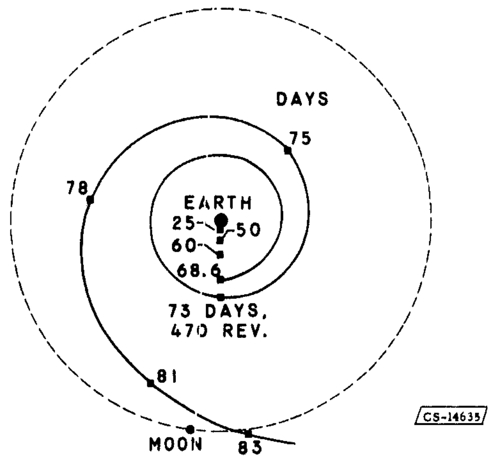
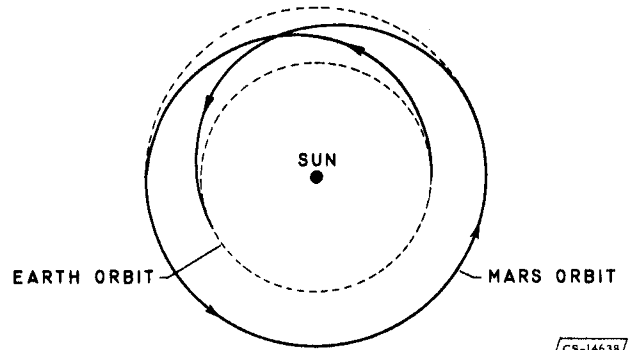


Figure 1

FLIGHT PATH TO MARS



TIME IN DAYS FOR MARS JOURNEY

SYSTEM	ESCAPE FROM EARTH ORBIT	COAST	DESCENT TO MARS ORBIT	WAIT	TOTAL
LOW THRUST ($F/W_0 = 10^{-4}$)	127	268	85	245	1205
IMPULSE ROCKET	–	268	–	415	951

Figure 2

EFFECT OF POWERPLANT WEIGHT ON PAYLOAD AND PROPULSION TIME FOR OPTIMUM SPECIFIC IMPULSE

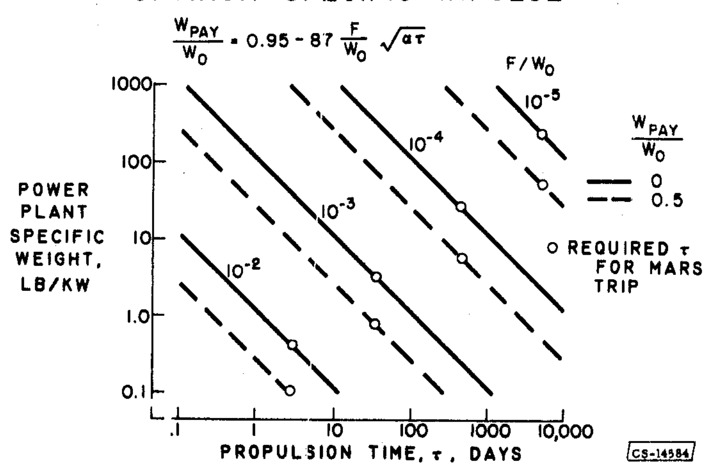


Figure 3

EFFECT OF NON-OPTIMUM SPECIFIC IMPULSE ON PAYLOAD FOR MARS TRIP

$F/W_0 = 10^{-4}$

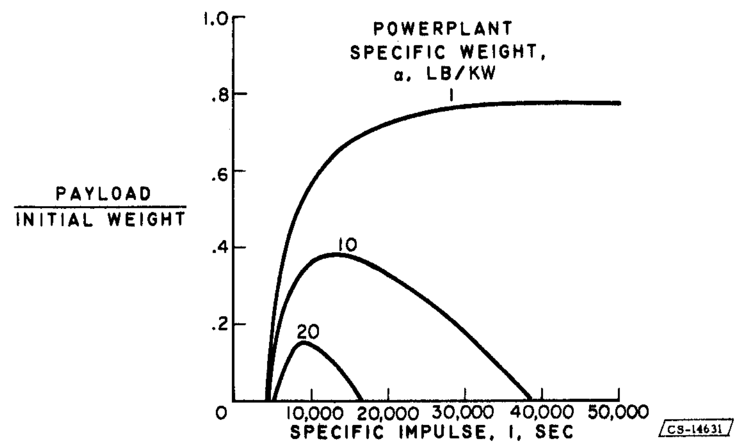


Figure 4

COMPARISON OF WEIGHT POWER RATIOS

(FOR COMMERCIAL CHEMICAL BATTERIES AT VARIOUS LOAD TIMES)

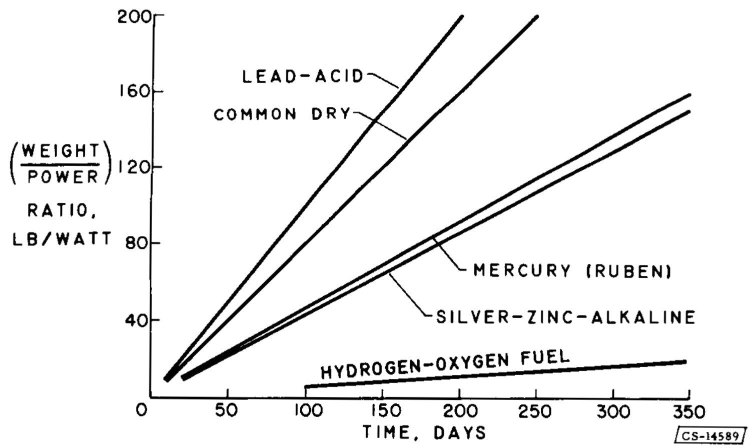


Figure 5

LOW POWER ELECTRICAL SOURCE

CONTINUOUS OPERATION P_O^{210} ENERGY SOURCE

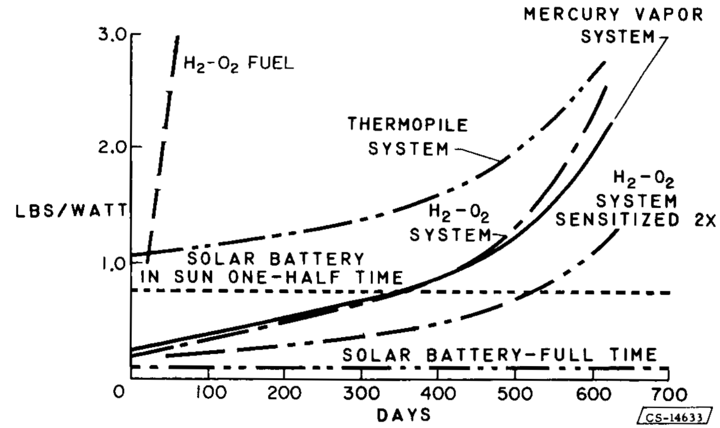


Figure 6

H_2-O_2 RADIOISOTOPE POWER SUPPLY

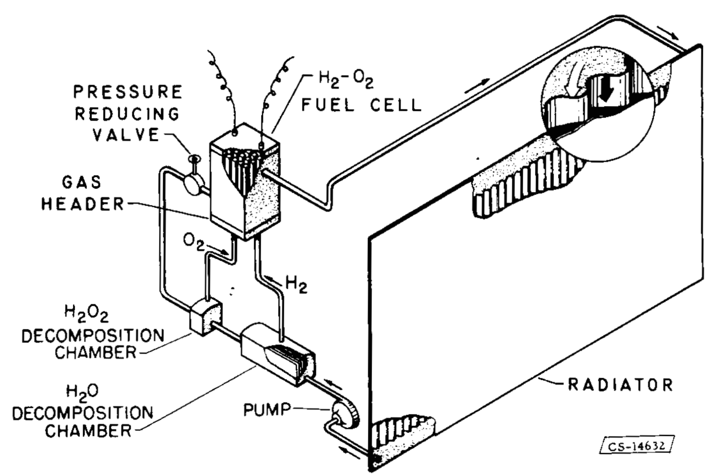


Figure 7

4793

SIMPLIFIED CYCLE ARRANGEMENT
SINGLE LOOP

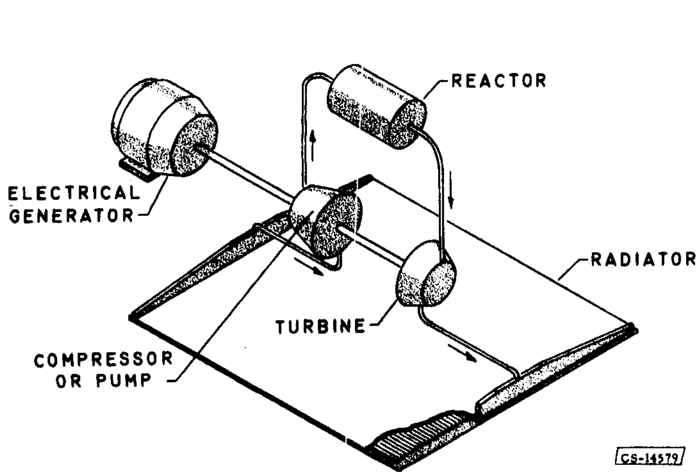


Figure 8(a)

SIMPLIFIED CYCLE ARRANGEMENT
TWO LOOP

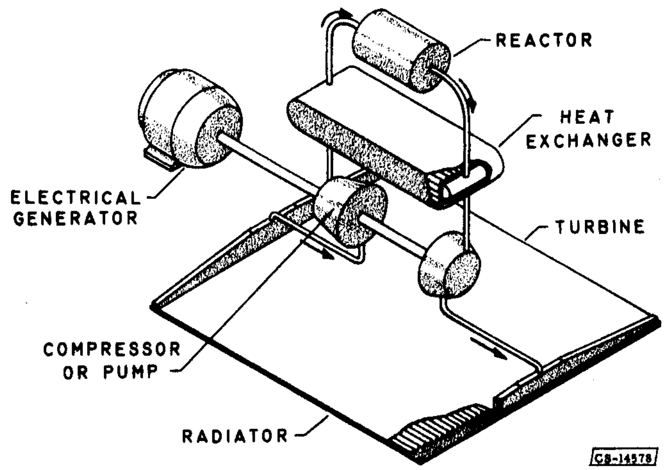


Figure 8(b)

RADIATOR AREA
TURBINE INLET TEMPERATURE, 2040 °F
$\eta_C = \eta_T = 0.8$

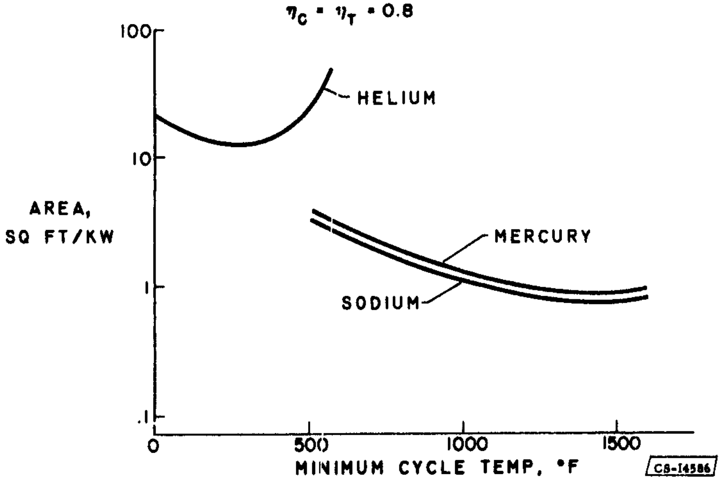


Figure 9

SCHEMATIC SODIUM-VAPOR SYSTEM

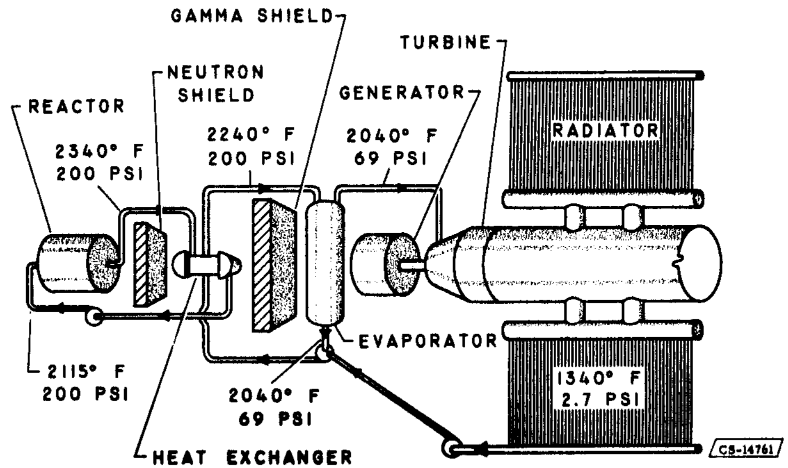


Figure 10

WEIGHT OF POWER SUPPLY

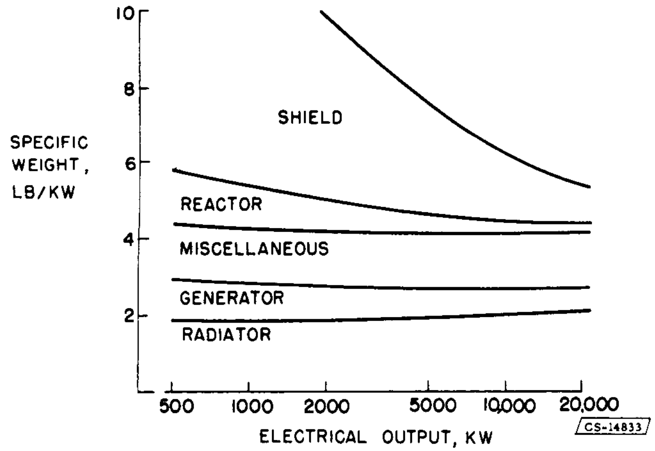


Figure 11

HYPOTHETICAL SPACE VEHICLE

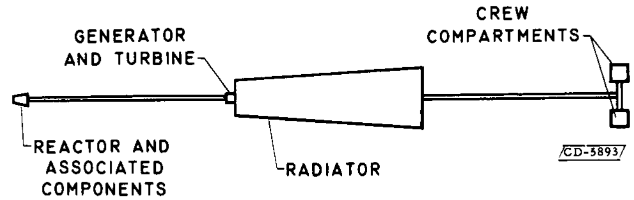


Figure 12

REACTOR END OF POWER PLANT

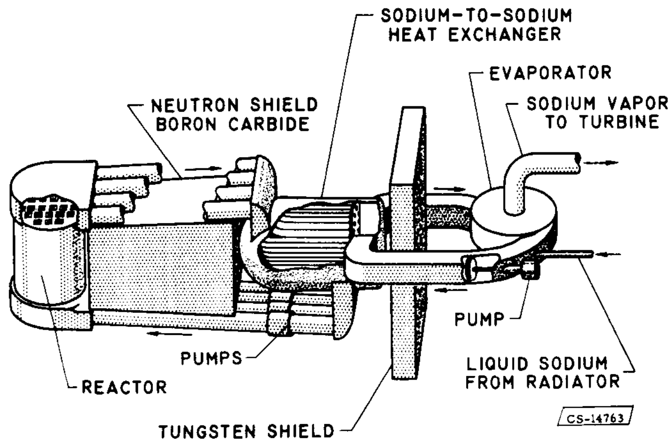


Figure 13(a)

TURBINE END OF POWER PLANT

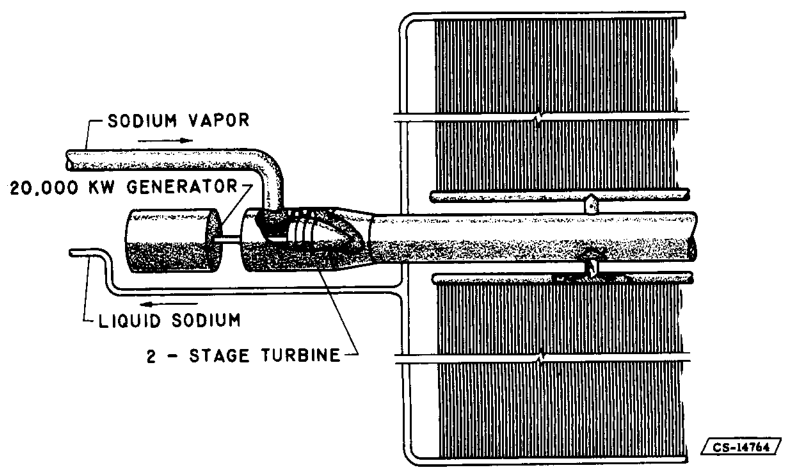


Figure 13(b)

SOLAR TURBO-ELECTRIC POWER SUPPLY

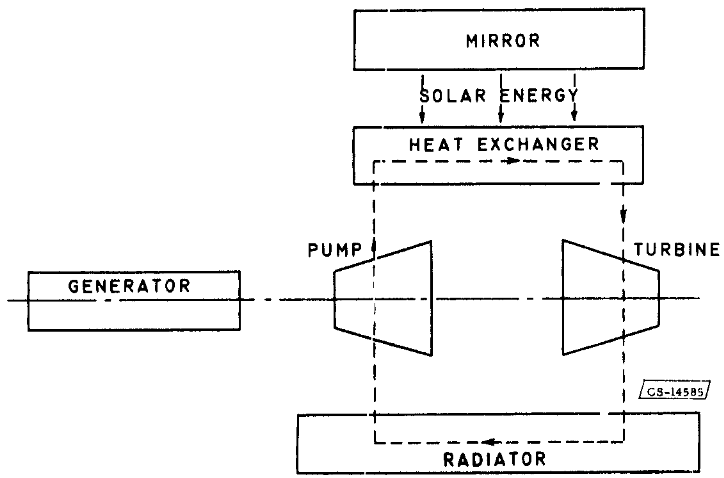


Figure 14(a)

SOLAR TURBO – ELECTRIC POWER SUPPLY

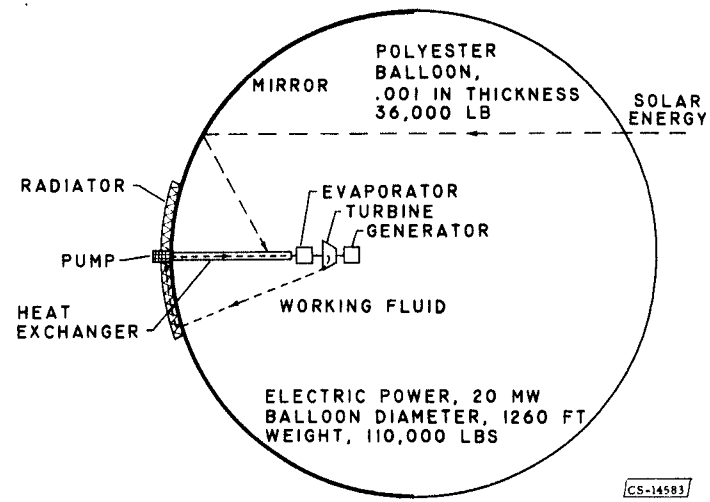


Figure 14(b)

ESTIMATED WEIGHT OF ELECTRIC POWER GENERATORS

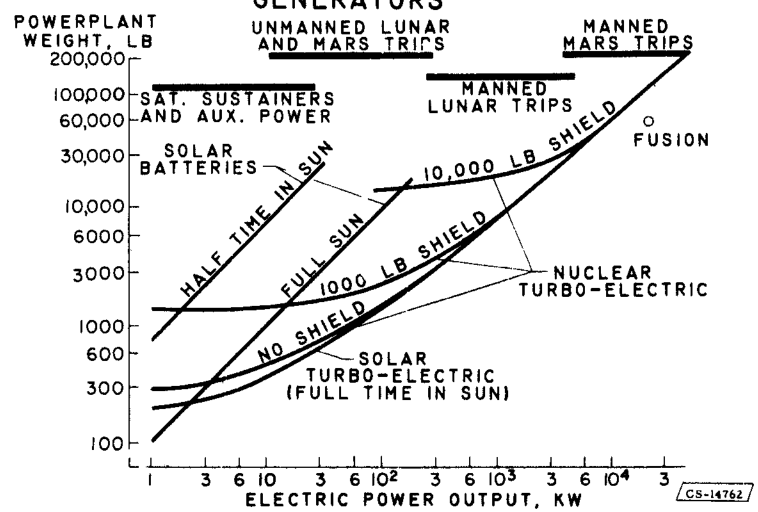


Figure 15

ARC-JET PROPULSION SYSTEM

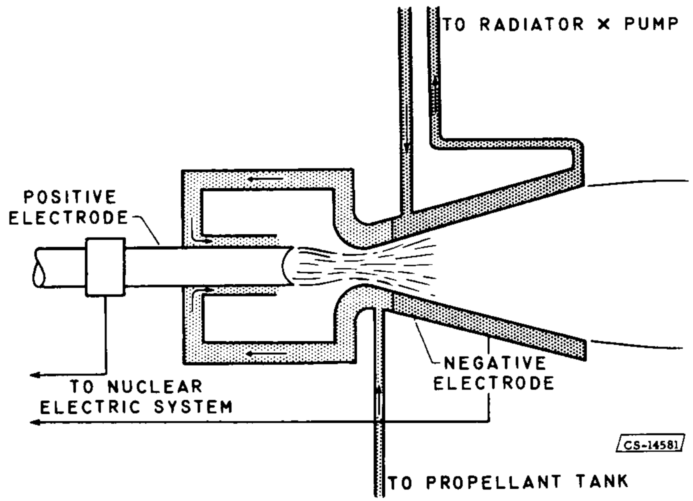


Figure 16

ARC CHAMBER PERFORMANCE

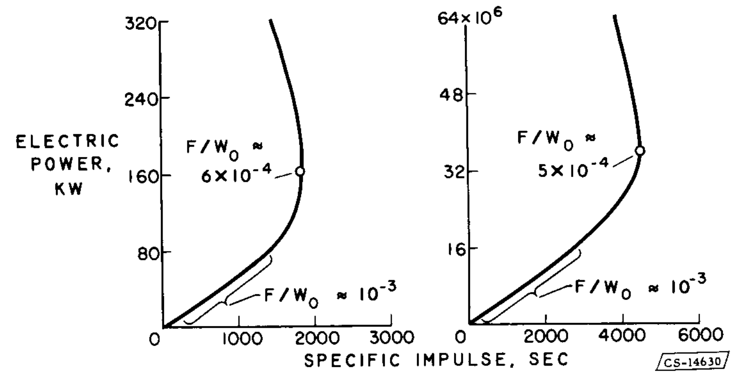


Figure 17

COMPONENTS OF ION THRUST SYSTEM

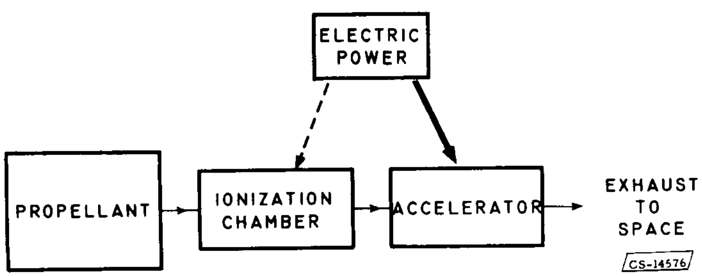


Figure 18

ION AND ELECTRON SOURCE (STUHLINGER)

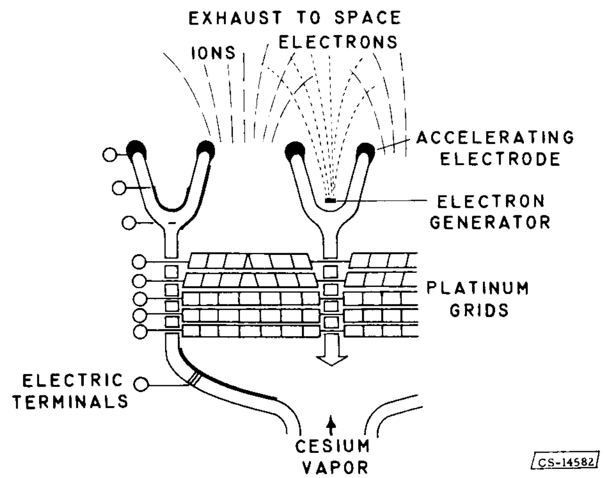


Figure 19

BOSTICK'S PLASMA ACCELERATOR

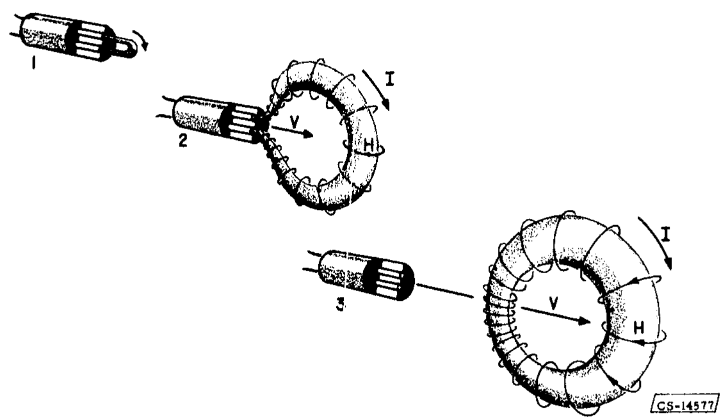


Figure 20

PLASMA 0.5 MICROSECONDS AFTER FIRING

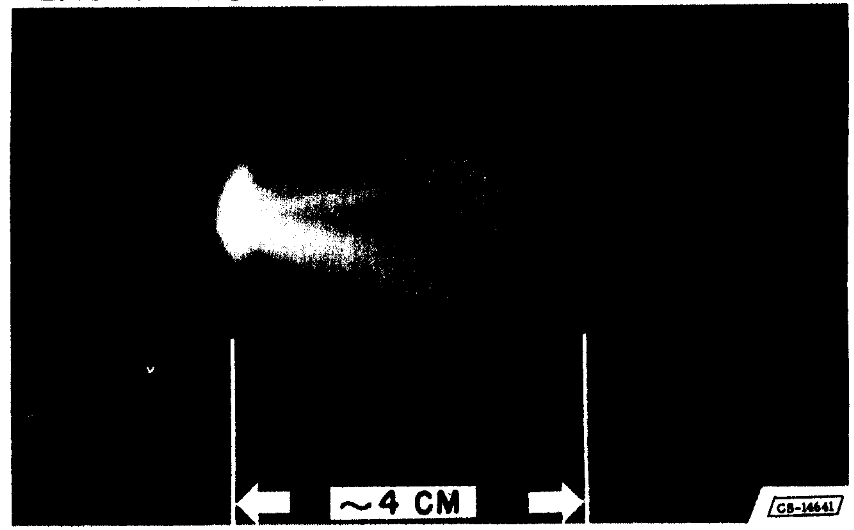
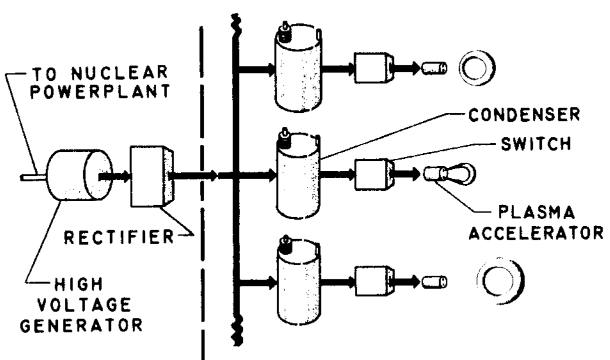


Figure 21

PLASMA ACCELERATOR PROPULSION SYSTEM



MARS JOURNEY

ACCELERATOR EFFICIENCY	OPTIMUM SPECIFIC IMPULSE	WEIGHT OF POWERPLANT AND PROPELLANT PER LB OF THRUST
0.40	11,000	7000
0.80	15,000	5000

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Figure 22

PHOTON SAIL

THRUST/SQ FT	2×10^{-7}
WEIGHT/SQ FT (t = 0.0005")	3×10^{-3}
THRUST/WEIGHT (IDEAL)	7×10^{-5}

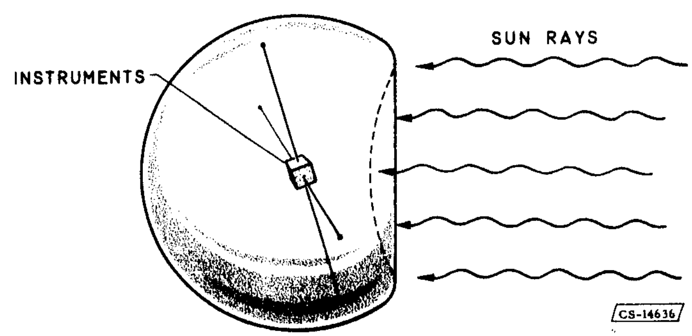


Figure 23

RADIOISOTOPE SAIL

THRUST/SQ FT 1×10^{-6}
WEIGHT/SQ FT (t = 0.0012") 9×10^{-3}
THRUST/WEIGHT (IDEAL) 1×10^{-4}

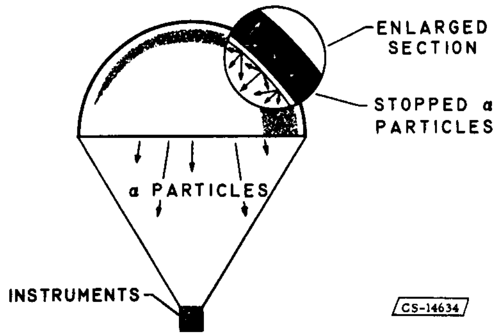


Figure 24

COMPARISON OF ENERGY AVAILABLE WITH ENERGY REQUIRED

IN IONOSPHERE ORBITS AT VARIOUS ALTITUDES

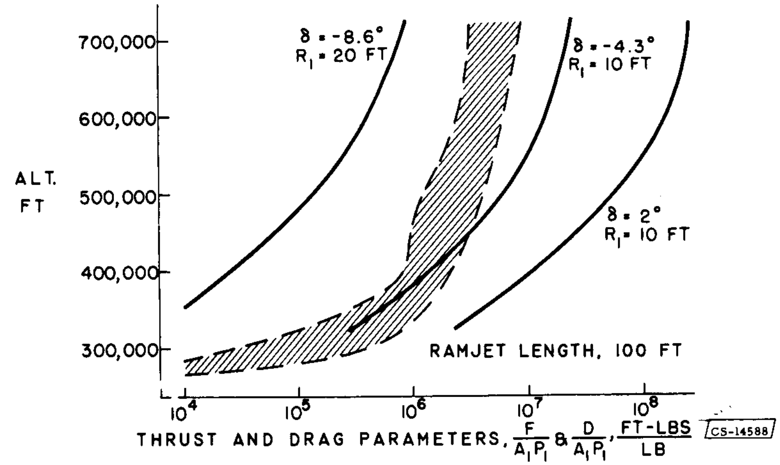
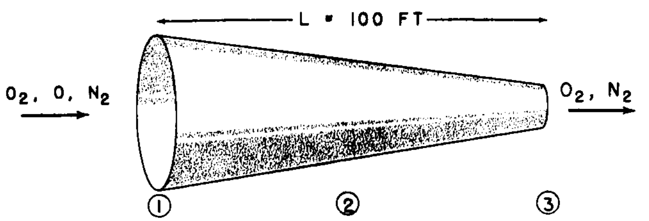


Figure 25

ONE POSSIBLE IONOSPHERE RAMJET ORBITING AT 328,000 FEET



D_1 = 40 FT D_2 = 0.46 FT D_3 = 10 FT
M_1 = 25.8 M_2 = 7.5 M_3 = 9.5

OVERALL CYCLE EFFICIENCY η = 22%
THRUST, F = 9.28 LBS
DRAG, D = .35 LBS

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Figure 26

THERMODYNAMIC CYCLE FOR IONOSPHERE RAMJET ORBITING AT 328,000 FEET

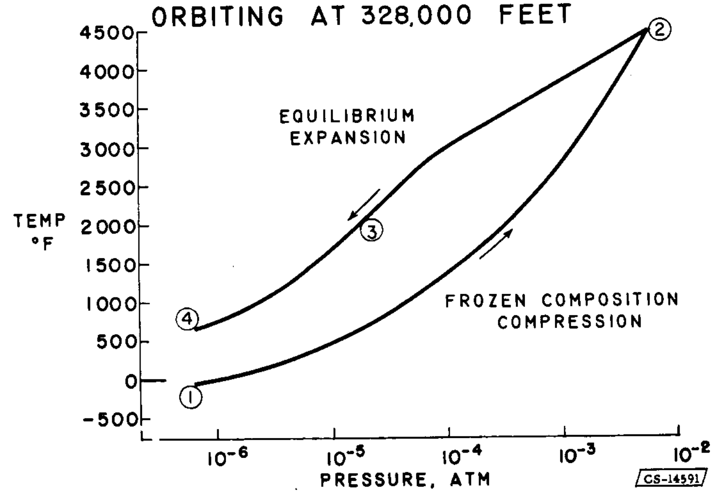

Figure 27

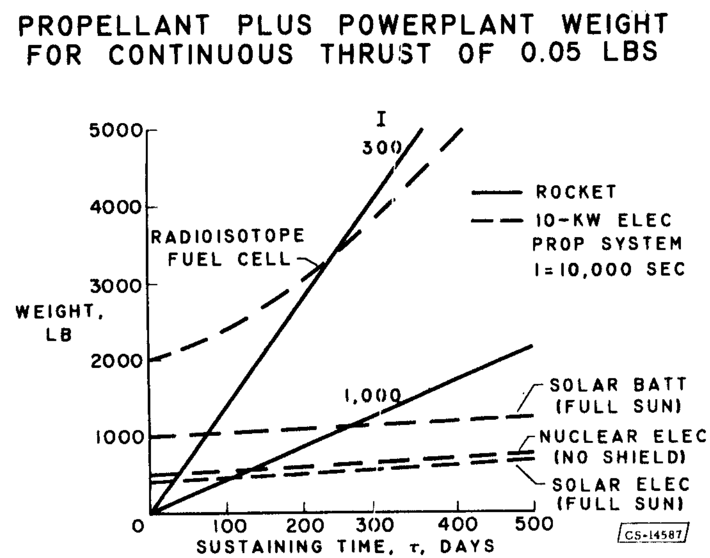

Figure 28

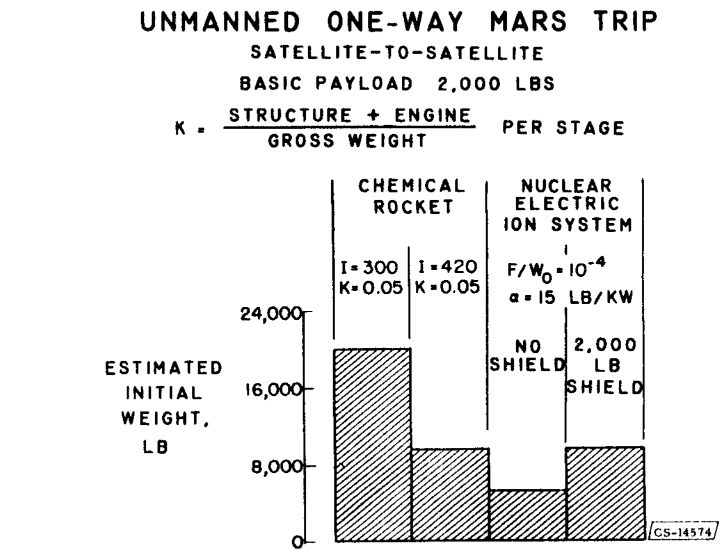

Figure 29

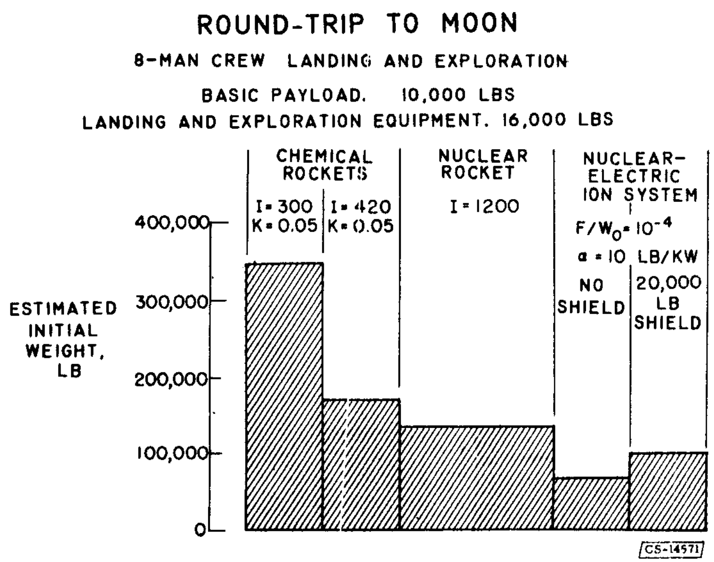

Figure 30

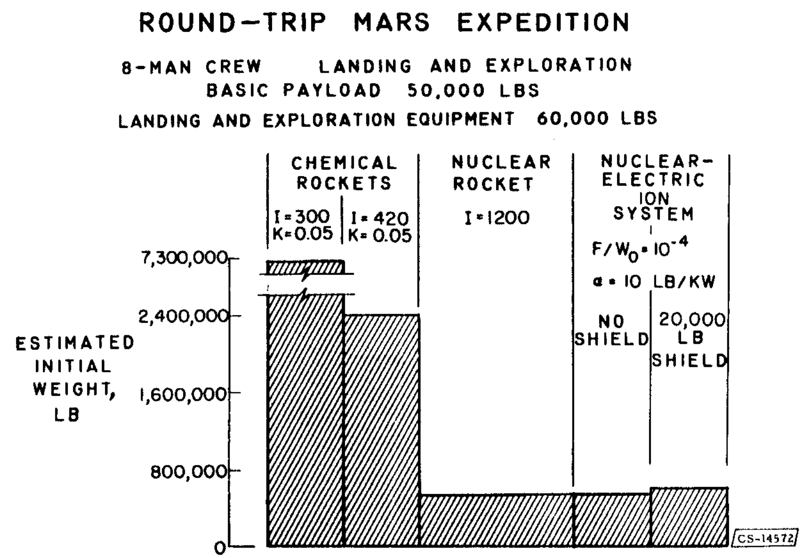

Figure 31